NASA Contractor Report 204139

Optical Closed-Loop Flight Control Demonstration

Fly-By-Light Aircraft Closed Loop Test (FACT) Program

Bradley L. Kessler
The Boeing Company
St. Louis, Missouri

and

Fly-By-Light Installation and Test (FIT) Program

Michael F. Wanamaker
The Boeing Company
Long Beach, California

August 1997

Prepared for
Lewis Research Center
Under Contract NAS3-25965

National Aeronautics and
Space Administration

Optical Closed–Loop Flight Control Demonstration

Contains Two Programs:

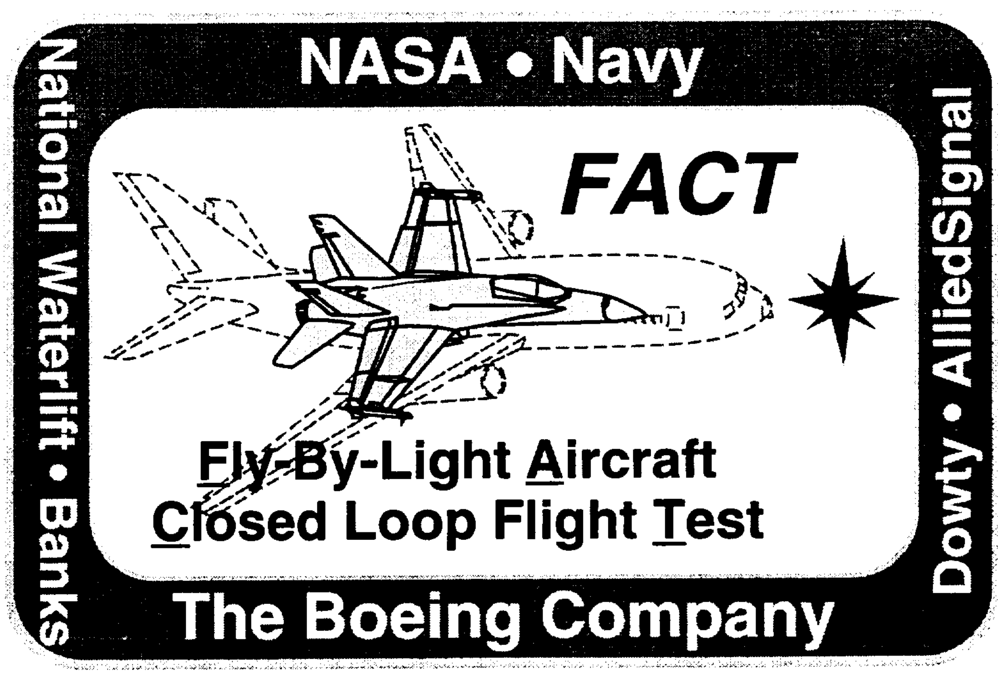
Fly–By–Light Aircraft Closed Loop Test (FACT) Program

Bradley L. Kessler
McDonnell Aircraft and Missile Systems
P.O. Box 516
St. Louis, MO 63166

and

Fly–By–Light Installation and Test (FIT) Program

Michael F. Wanamaker
McDonnell Aircraft and Missile Systems
2401 East Wardlow Road
Long Beach, CA 90807



GP71485001.cvs

TABLE OF CONTENTS

Page

LIST OF FIGURES AND TABLES

ABBREVIATIONS AND ACRONYMS

AOC	Active Optical Contact
BIT	Built–In–Test
C	degrees Celsius
CAS	Control Augmentation System
CCD	Charged Coupled Device
CE	Conducted Emission
CS	Conducted Susceptibility
DSP	Digital Signal Processor
EHV	Electro–Hydraulic Valve
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EOA	Electro–Optic Architecture
FACT	Fly–By–Light Aircraft Closed–Loop Test program
FBL	Fly–By–Light
FBW	Fly–By–Wire
FCC	Flight Control Computer
FCS	Flight Control System
FCES	Flight Control Electronic Set
FIT	Fly–by–light Installation and Test program
FLASH	Fly–by–Light Advanced Systems Hardware program
FLOAT	The Fly–by–Light Optical Aileron Trim program
FOCSI	Fiber Optic Control Systems Integration program
FRU	Feedforward Remote Units
FTCP	Flight Test Control Panel
g	force of Gravity
IBIT	Initiated Built–In–Test
ICU	Interface Converter Units
IM	Interface Module
LRU	Line Replaceable Unit
LVDT	Linear Variable Differential Transformer
MIL–STD	Military Standard
NASA	National Aeronautics and Space Administration
NAVMAT	Naval Material Command Document
NAWC	Naval Air Warfare Center
PC	Personal Computer
PMECH	Pitch Mechanical
RE	Radiated Emission
RF	Radio Frequency
rms	Root Mean Square
ROC	Reliable Optic Connector
RS	Radiated Susceptibility
SEM	Standard Electronic Module
SHARP	Navy Standard Hardware Acquisition and Reliability Program
SRA	Systems Research Aircraft
UUT	Unit Under Test
WDM	Wave Division Multiplexing

THIS PAGE INTENTIONALLY LEFT BLANK

FACT SUMMARY

The Optical Closed-Loop Flight Control Demonstration contains two programs: Fly-By-Light Aircraft Closed-Loop Test (FACT) and Fly-by-light Installation and Test (FIT). Both program final reports are included in this report.

The objective of the Fly-By-Light Aircraft Closed-Loop Test (FACT) program is to demonstrate an in-flight optical closed loop control system equivalent to a production Fly-By-Wire (FBW) system for a rudder control surface. The FACT system has been designed, developed, and tested with results showing a rugged, well performing system ready for flight tests on the NASA-Dryden F/A 18 Systems Research Aircraft. The FACT program was sponsored by NASA-Lewis Research Center and developed by McDonnell Aircraft and Missile Systems. This final report describes the FACT system architecture, development, and test up to delivery to NASA-Dryden.

The FACT system consists of interface computers in composite chassis, remote terminals, unmodified flight control computers (FCCs), and modified rudder actuators. The actuators were modified by replacing main ram FBW feedback position sensors, linear variable differential transformers (LVDTs), with optic position sensors. The interface computer receives control surface commands from the flight control computer, converts the electrical command signals to optic signals, and sends the optic signal to the remote terminals. The remote terminal converts the optic signal to electric to control the actuator, monitors the electric current through the actuator, and sends an optic signal of the current to the interface computers for error monitoring. The interface computer also decodes the actuator optic position sensor signal, converts the optic signal to electric, sends the actuator position to the flight control computer, and checks for errors. The conversions between electric and optic signals minimized costs by preventing major modifications and revalidation of existing flight control hardware.

Several tests verified FACT system readiness for flight test. Flight simulations determined the severity of failure modes. Acceptance tests verified the performance of the modified actuators. Component tests verified the proper operation of each low-level function in the avionics. Vibration and temperature stress screening of completed modules verified the workmanship of assembly and the quality of electronic components. Vibration, temperature and altitude, and electromagnetic compatibility airworthiness tests verified the ability of the system to survive the military aircraft environment. System tests provided an aircraft-like environment and verified the performance of the FACT system is equivalent to the performance of the production fly-by-wire system.

The FACT system test results from component tests, system tests, and airworthiness tests show the FACT fly-by-light system performs well, performs similarly to the production fly-by-wire system, and will survive the aircraft flight test environment. The FACT fly-by-light system is ready for flight test.

FIT SUMMARY

The objective of the Fly-by-light Installation and Test (FIT) program is to investigate fiber cable installation, repair, and signal transmission in an aircraft environment.

The fiber and electrical cables were installed in convoluted tubing by pulling the conductors through the tubing. It was very difficult to pull the conductors through the tubing due to the many bends in the routing. Also, the conductors braided together as they were being pulled through the tubing. The braiding made it impossible to remove any one conductor from the conduit or insert another conductor. An alternative method for repair or installation is needed.

Several cable repairs were accomplished on the aircraft. The ease of repair was dependent upon the location and accessibility of the break. The single-channel splice and visual fault finder were successfully used in the repairs.

The active optical contact (AOC) showed acceptable performance. The AOC prevents the fiber's end face from becoming contaminated by dirt and debris when a protective dust cover is not installed, but the AOC makes it difficult to connect fiber-optic test equipment to the optical cable harness.

An EMI conducted susceptibility problem in the Optical Interface Unit prevented any meaningful analysis of the optical and electrical signals. This problem was not corrected by the time the program was terminated. Likewise, the ground testing and scheduled maintenance activities were not performed.

THIS PAGE INTENTIONALLY LEFT BLANK

1. FIT Introduction

The objective of Fly-By-Light Installation and Test (FIT) is to ensure the compatibility of Fly-By-Light (FBL) systems with commercial aircraft installation requirements and practices. In order to meet this objective, a cable harness with optic fibers and electric wires was fabricated and installed aboard NASA's Systems Research Aircraft (SRA).

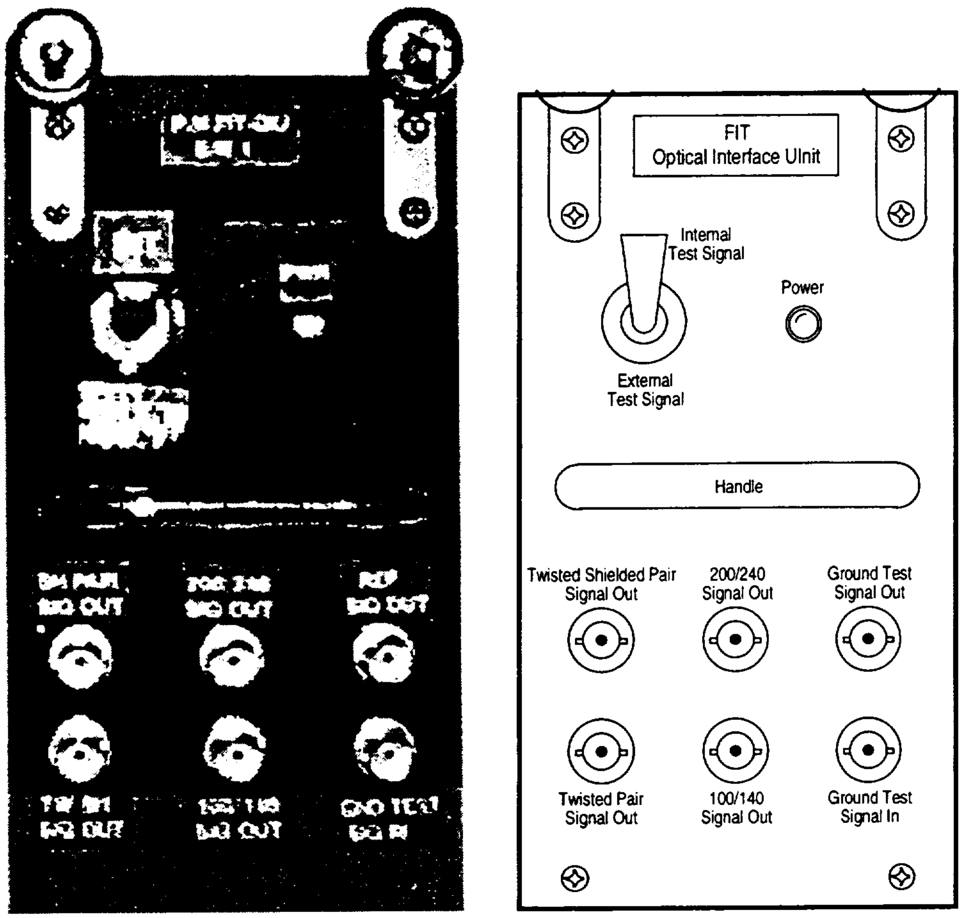
FIT program objectives were:

1. Correlate all optical signal quality disturbances with temperature, vibration, cable bending, aircraft acceleration and maneuvers, and standard installation and maintenance practices.
2. Compare the optical results with the electrical results.
3. Analyze any optical signal quality disturbances to determine avoidance methods.
4. Recommend cable handling, routing and content to avoid causative factors in future installation.
5. Test and evaluate the single-channel splice, Siecor Camsplice™, active optical contact (AOC) and cable repair techniques.
6. Perform system ground testing with externally generated test signals of varying frequency.
7. Perform cable splicing and/or removal and connector unmate/remate cycles during aircraft down times.

THIS PAGE INTENTIONALLY LEFT BLANK

2. FIT System description

In order to accomplish the project objectives, an optical interface unit (OIU) was constructed. This OIU, installed in bay 13L, will generate the in-flight test signals for both the optical cable and copper wires. In addition, the OIU will allow for externally generated signals to be injected into the cable harness. The returned signals are sent to the on-board data recorder and stored until they are down-loaded for analysis.



GP71485060.cvs

Figure 1. FIT Optical Interface Unit

The cable harness consists of two electrical conductors (#20 AWG twisted pair wire and #20 AWG twisted, shielded, jacketed pair wire) and two fiber optic cables (100/140 and 200/240). A common signal is injected into these conductors by the OIU for flight and ground testing. The cable harness is routed from bay 13L, up through the turtleback, to the right engine compartment, to the right wheel well, to the right wingfold, back to the turtleback and returns to bay 13L. The harness is approximately 130 feet long and is enclosed in a fluorinated ethylene propylene (FEP) conduit.

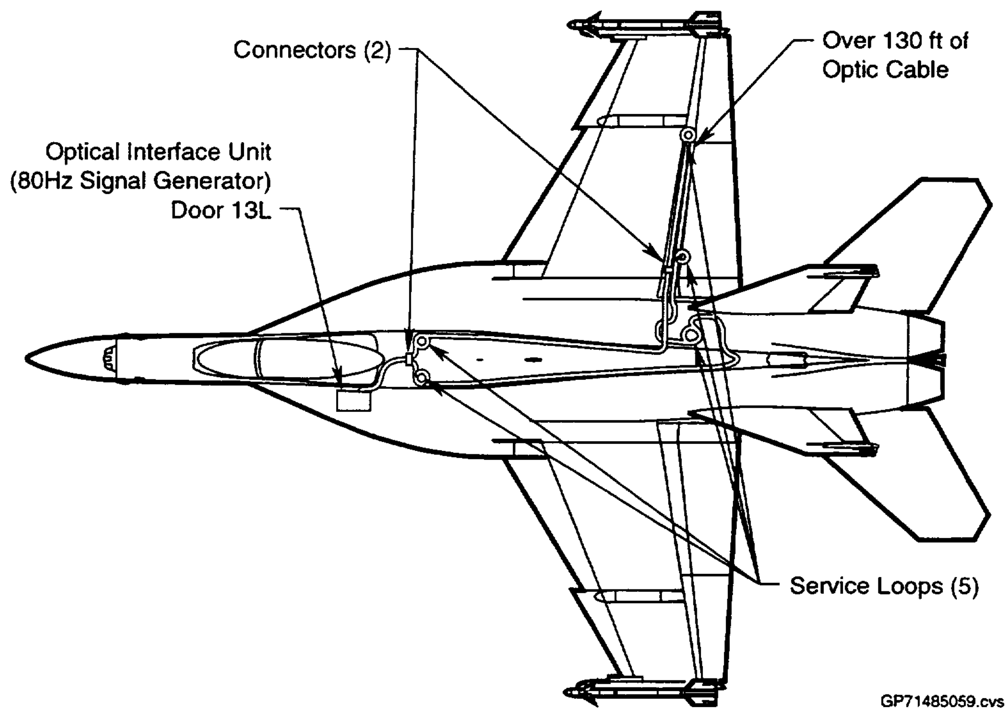


Figure 2. FIT Equipment Installed on Aircraft

3. FIT Tests Planned, Performed, Test Results, and Discussion

FIT tests were planned as follows before the program was halted due to a stop-work order. The testing will be conducted on a "non-interference" basis during other test flights. Data will be recorded during each flight for post-flight reduction and analysis. The cable harness and OIU will be left aboard the SRA for 100 flight hours. Scheduled maintenance such as connector unmate/mate cycles and cable harness repair, will be performed during aircraft down time. Field repair techniques such as cable repair and replacement will be evaluated. In addition, signal quality tests at various frequencies will be performed, on the ground with engines running, to determine the effects of EMI/RFI.

The flight test phase of the program has begun, and test flight data has been taken. The data indicated that there was interference on all four channels. Troubleshooting of the OIU and harness indicated that an EMI conducted susceptibility problem exists which is attributed to the electrical signal wiring and aircraft's electrical power system. No interference exists when the OIU is connected to a laboratory power supply. Some interference exists when the OIU is connected to the aircraft power supply. More interference exists when electrical signal cables are connected to the OIU. Efforts to troubleshoot the system were initiated and then halted due to a stop-work order placed on the program. The EMI problem needs to be resolved before optic signal quality analysis can be performed.

THIS PAGE INTENTIONALLY LEFT BLANK

4. FIT Conclusions

During cable harness installation, it was very difficult to pull the four conductors through the conduit due to the many bends in the conduit routing. Also, the four conductors braided together as they were being pulled through the conduit. This braiding effect made it impossible to remove any one conductor from the conduit or insert another conductor since doing so may damage a conductor already in the conduit. It is therefore concluded that removing a conductor from the conduit is impractical and an alternative method for repair is needed.

During harness installation, the 200/240 cable broke. The visual fault finder (VFF) was used to find the location of the break(s). The VFF, which emits visible (red) laser light that is coupled to the optical cable under test, worked well with the optical cable's purple jacket. At least one cable break was located using the VFF. If a break is present, the laser light will illuminate the purple jacket, indicating the location of the break. However, the VFF will only show the first break in the cable. One method to determine if there are multiple breaks is to connect the VFF to the other end of the cable. If the same area lights up, there is only the one break. If that spot does not light up, there is at least one additional break in the cable. If there are no breaks, the laser light will emerge at the other end of the cable. For VFF to be effective, complete visual access to the entire cable under test is required.

Several cable repairs were accomplished on the aircraft. The ease of reparability was dependent upon the location and accessibility of the break. The single-channel splice was used on the 200/240 cable. This splice uses standard MIL-C-83723 optical termini and mates them in a single channel connector. No special tools were required to accomplish this splice. Only the standard termination equipment was needed.

The Siecor Camsplice™ could not be used due to the 240μm cladding diameter of the fiber. This splice was not designed for this large a fiber. This splice uses an index-matching gel that is well protected in the splice and does not attract dirt or debris that can increase optical attenuation.

The active optical contact (AOC) showed acceptable performance. The advantage of the AOC is that it provides an electrical interface, as opposed to an optical interface, at the line replaceable unit (LRU) connector. This prevents the fiber's end face from becoming contaminated by dirt and debris when a protective dust cover is not installed. One drawback to the AOC is that it makes it difficult to connect fiber-optic test equipment to the optical cable harness.

The EMI/RFI problem prevented any meaningful analysis of the optical and electrical signals. This problem was not corrected by the time the program was terminated. Likewise, the ground testing and scheduled maintenance activities were not performed.

THIS PAGE INTENTIONALLY LEFT BLANK

5. FACT INTRODUCTION

5.1 Background

Recent government and industry activities aimed at promoting Fly-By-Light (FBL) to standard practice are illustrated in Figure 3. These programs have advanced FBL by developing FBL components and systems. The Fiber Optic Control Systems Integration (FOCSI) program repackaged existing optic sensing technologies, developed a single electro-optic architecture to decode the various technologies, and monitored open loop optical sensors in flight. The Fly-by-Light Advanced Systems Hardware (FLASH) program developed optically interfaced inertial measurement units, optically interfaced actuators and flight control computers, and an optically based distributed architecture vehicle management system. The Fly-by-Light Optical Aileron Trim (FLOAT) program developed optic trim actuators and optic data bus for a commercial aircraft production system for the benefits of reduced weight and increased reliability. The Fly-By-Light Aircraft Closed-Loop Test (FACT) program developed optic sensors and an electro-optic architecture (both were improvements over the FOCSI program), mounted the optic sensors inside the flight control actuators, and demonstrated optical closed loop control of a flight control actuator. The FACT program is a key link in FBL development since it provides FBL performance equivalent to a production Fly-By-Wire (FBW) system in an environment applicable to commercial and military aircraft.

Hardware and lessons learned in the FACT program were used in the FLASH program and the FLOAT program. The FLASH program used the FACT optic sensors and EOA module to provide optic position feedback for the pilot stick in a closed loop flight simulation demonstration that used control surface actuators, flight control computers, and a six degree of freedom aircraft model. The FLOAT program used a modified FACT EOA module with custom software to decode optic sensors.

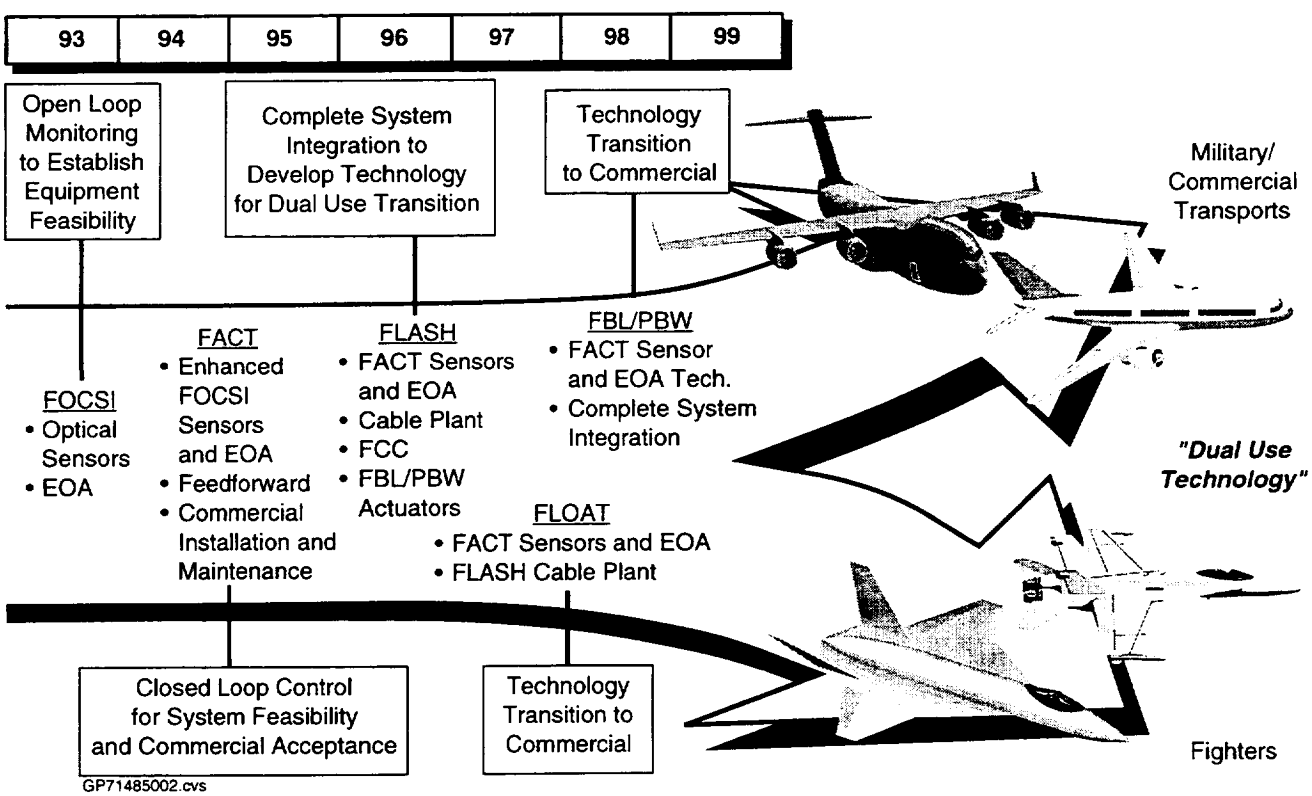


Figure 3. Fly-By-Light Programs Transition the Technology

The FBL system developed in the FACT program will be installed in the NASA Dryden F/A-18 Systems Research Aircraft (SRA). This selection was based on providing an aircraft that could be economically modified to FBL while also providing an environment that is relevant to both commercial and military applications. By integrating the FBL modifications into the existing FBW flight controls an affordable demonstration was achieved. The use of a military aircraft covers the complete spectrum of commercial and military environmental conditions as shown in Figure 4.

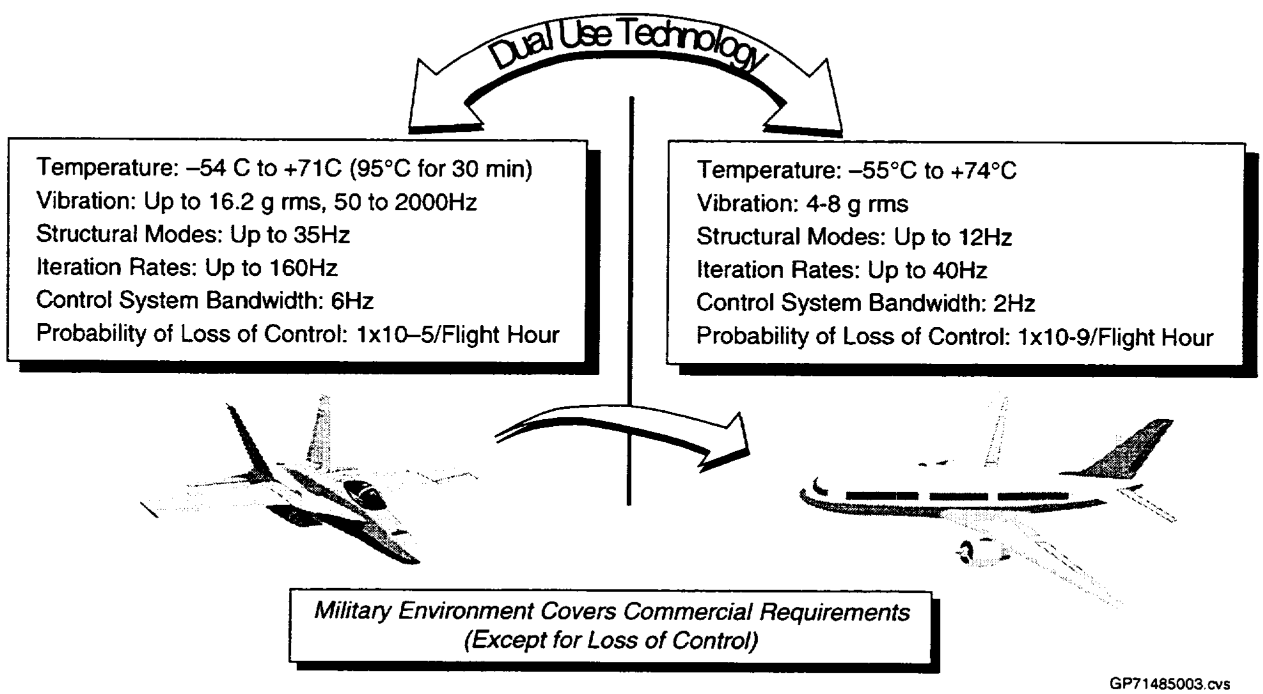


Figure 4. FBL Test on SRA is Operationally Relevant

5.2 Scope

The FACT system was to demonstrate Fly-By-Light control of a stabilator and rudder flight control surface, but problems in assembling the stabilator optic sensors prevented completion of the stabilator actuators in time to test the actuators and the FACT system with actuators. The result is the FACT system was designed and developed for a stabilator and rudder flight control surface, but the FACT system was integrated and tested only for the rudder control surface.

Control loop issues of update rate and data latency needed for performance equivalent to the Fly-By-Wire (FBW) production system were investigated in early testing that simulated in hardware the proposed architecture. Those test results were used as specifications for the electro-optic architecture (EOA) that decodes both rudder and stabilator optic sensors. The proposed architecture was again simulated but in software for flight simulation tests that investigated failure scenarios for the rudder and stabilator surfaces for various flight conditions. Those results helped to determine the FBW system was not needed as the FACT backup system and the standard aircraft backup system of stabilator mechanical control and rudder damping would suffice. The results were also summarized in the FACT hazards analysis report. The components of the FACT system were developed and each component tested for ruggedness and performance. Each rudder sensor and electronic module went through environmental stress screening and performance checks. After the sensors were installed in actuators, F/A-18 rudder acceptance tests were performed on each actuator. The electronic modules were integrated into the avionics boxes and tested, then the FACT system performance and ruggedness were tested. Component tests verified the performance of the optic avionics, system tests verified the performance of the system, and environmental airworthiness tests on one of each avionics box and rudder actuator validated the ruggedness of the optic avionics and modified actuators.

Key characteristics of the FACT flight test architecture that affected the scope of the program are summarized. The FACT system is full time FBL; there is no FBW backup. This meant the system needed to be robust and maintain all flight control error monitoring. The optical sensors are mounted inside the actuators, and the flight control redundancy is maintained (i.e., dual rudder and quad stabilator). This meant the optical sensors needed to be very rugged to survive a harsh environment and compact to fit two or four in a one inch diameter cylinder. No flight control computer (FCC) hardware or software modifications are allowed to avoid very costly revalidation of the FCCs. This meant the FACT system had to contain all optic interfaces, hardware to simulate existing interfaces, and provide error monitoring of equipment added to the flight control system. The electro-optic architecture (EOA) module is a Navy Standard Hardware Acquisition and Reliability Program (SHARP) common module. This meant the EOA had to meet stringent environmental conditions, and be flexible for other programs.

5.3 The Fact Team

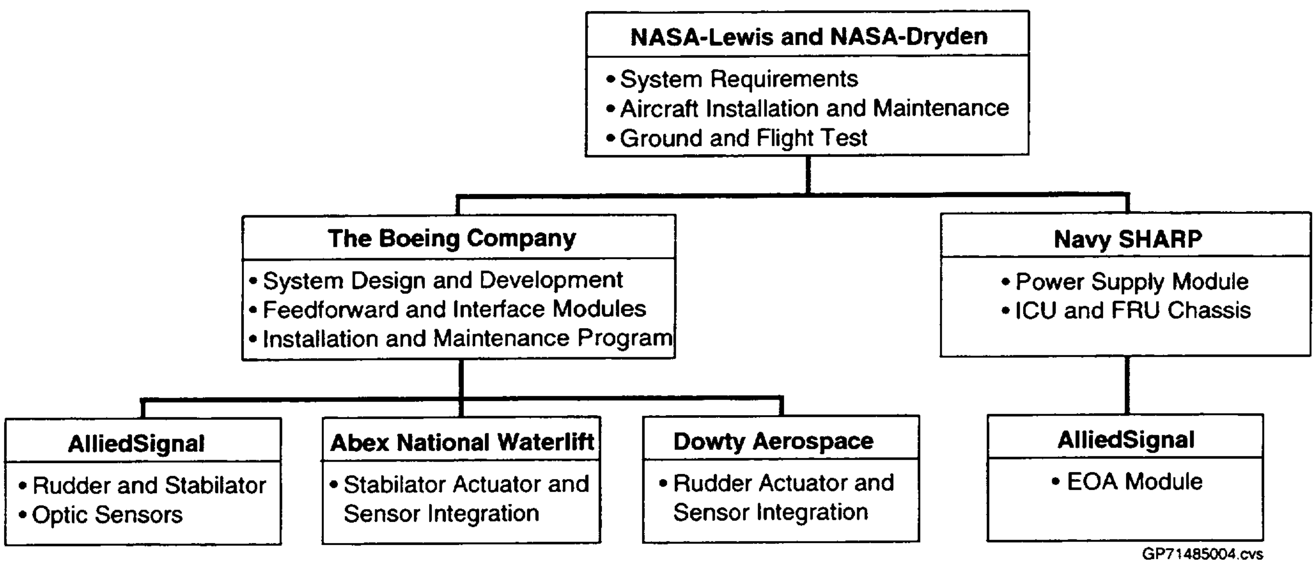
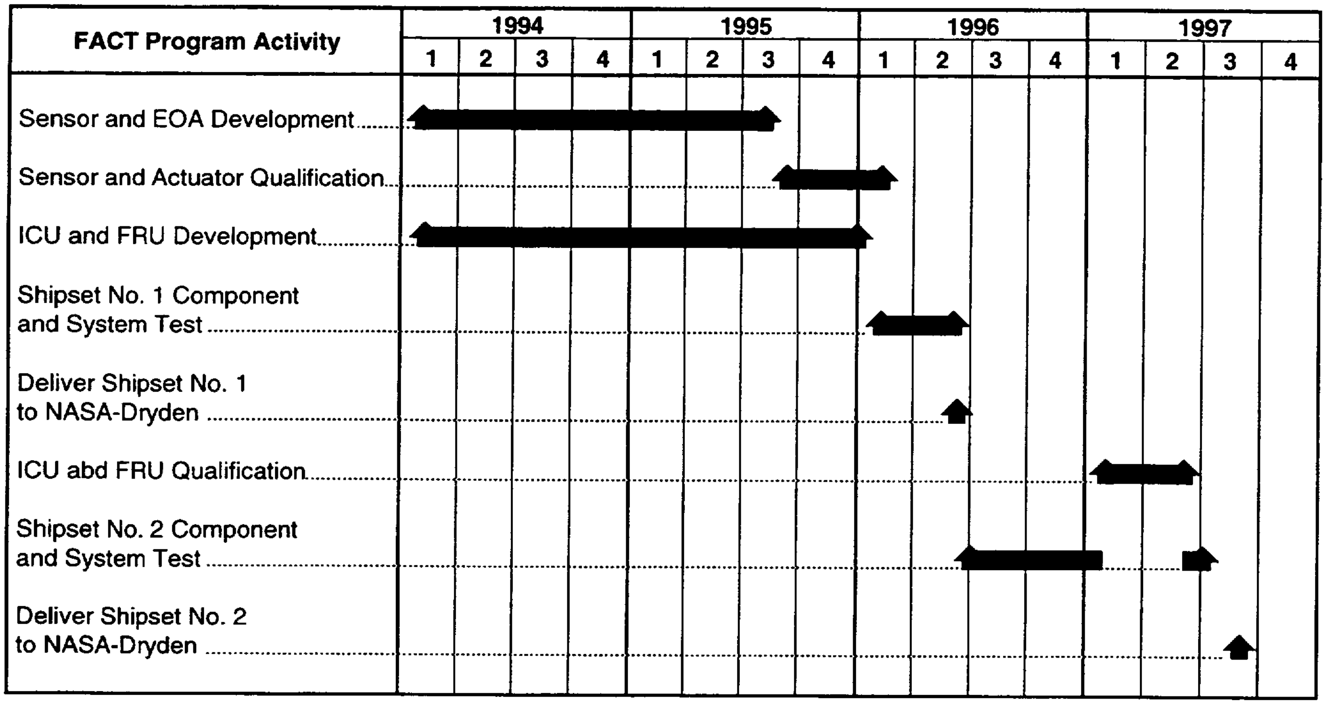


Figure 5. FACT Program Organization

The FACT program is a flight test program sponsored by NASA-Lewis Research Center and flight tested by NASA-Dryden Flight Research Center. The Navy's Standard Hardware Acquisition and Reliability Program (SHARP) supplied the ICU and FRU chassis, power supply modules, and EOA modules. Boeing's McDonnell Aircraft and Missile Systems is the system designer, developer, and integrator. Boeing designed and built the feedforward modules and interface module. Boeing also designed and built the FIT hardware and helped NASA-Dryden with the installation and test of the FIT system. AlliedSignal of South Bend, Indiana designed and built the EOA modules and the optical sensors for the rudder and the stabilator actuators. Dowty Aerospace supported the design of the rudder optic sensors, installed optical sensors into three rudder actuators, and performed acceptance tests. Abex National Waterlift supported the design of the stabilator optic sensors. Stabilator sensors were not installed in actuators due to difficulties in making the sensors.

5.4 FACT Program Schedule



GP71485005.cvs

Figure 6. FACT Program Schedule

6. FACT System description and development

This section describes the FACT system with the rudder and stabilator since the system was designed and developed for both actuators.

6.1 System Description

6.1.1 System Overview

The FACT system consists of the F/A-18 Flight Control Computers (FCCs), the modified left rudder and left stabilator actuators, avionic interface units to provide the optic and electrical signal interface at the FCCs and the actuators, fiber optic cabling, and cockpit controls. Figure 7 provides an overview of the system illustrating the new, modified, and unmodified equipment. In order to minimize cost, no modifications were made to the F/A-18 FCC, cockpit controls or flight control motion sensors. The modified and the new items for the FACT implementation indicated in Figure 7 retain the same level of redundancy of the F/A-18 production flight controls. The redundancy for the stabilator axis of control, Figure 8, indicates how the quadruplex redundancy is carried from the existing FCCs through the Interface Converter Units (ICUs) and optical cables to the Feedforward Remote Units (FRUs) and the optical sensors mounted in the flight control actuators.

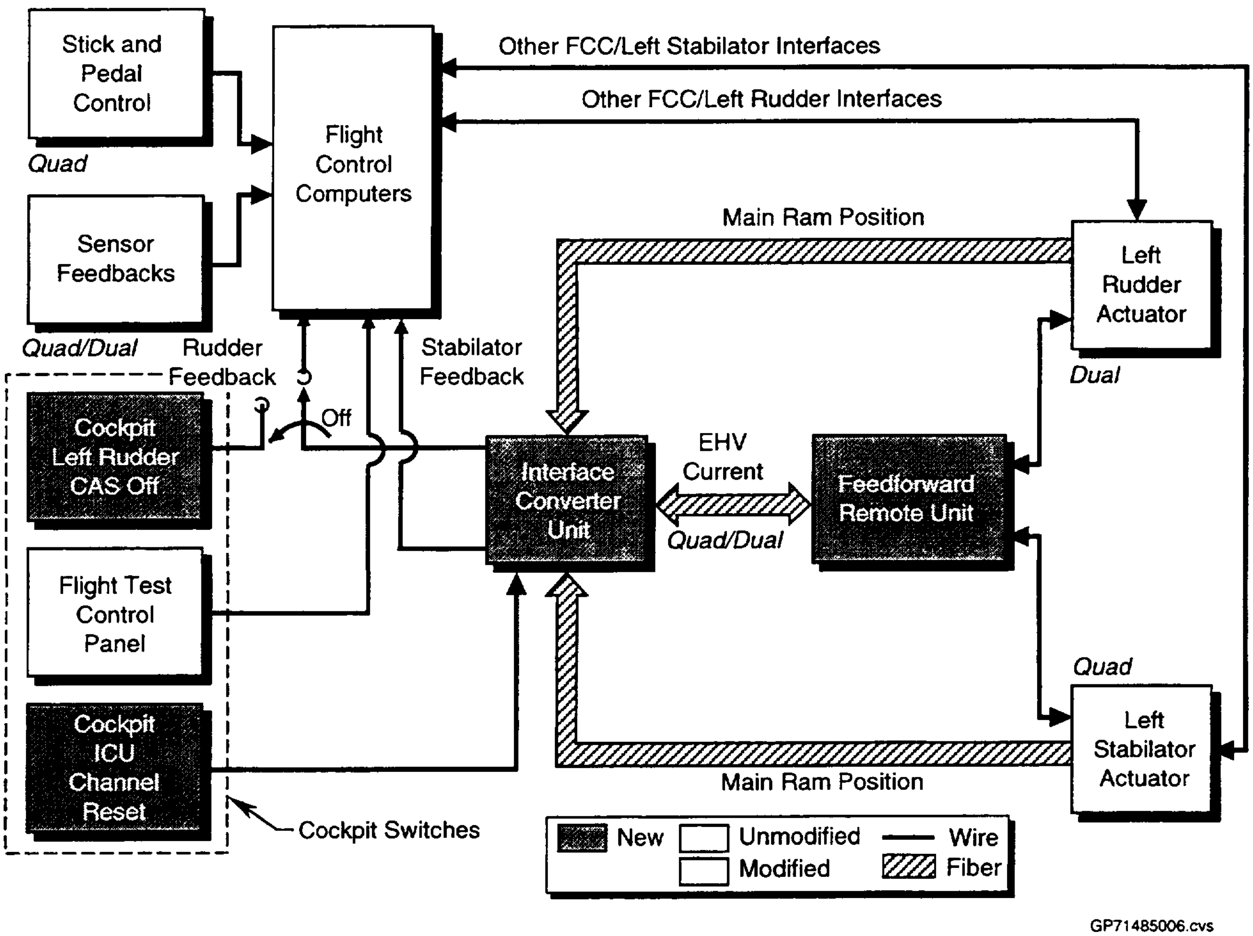


Figure 7. FACT System Overview

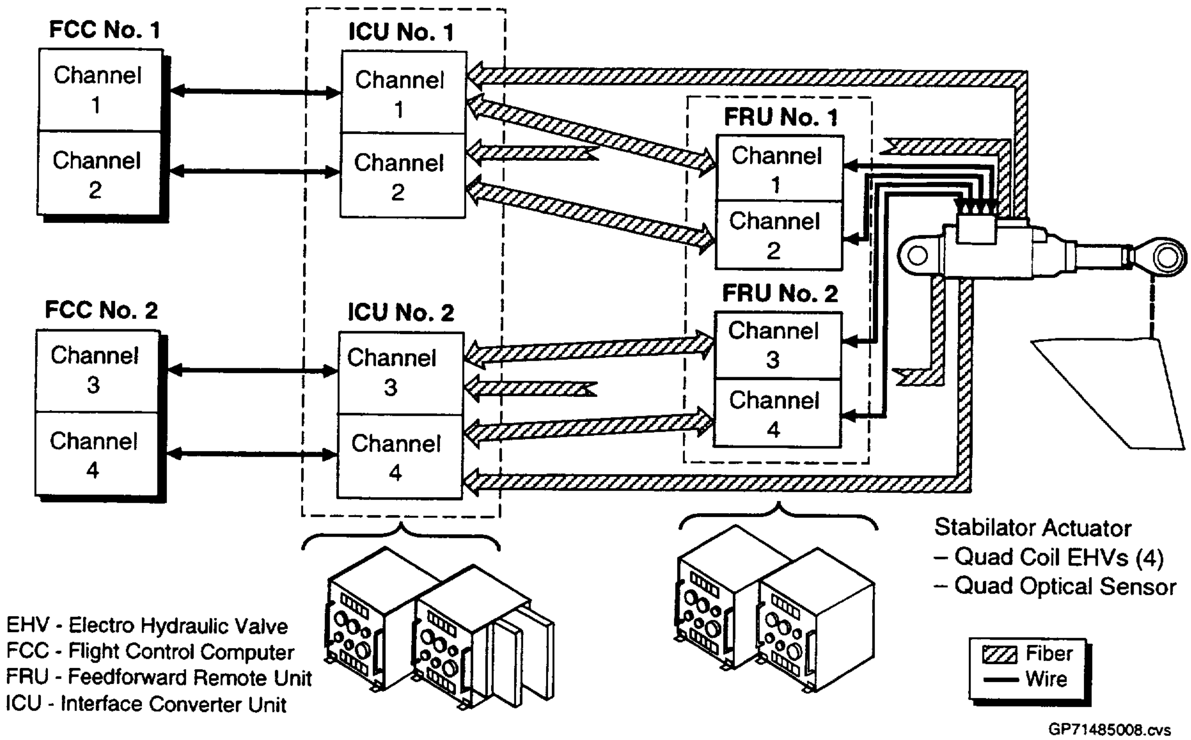


Figure 8. Quad Stabilator Axis Redundancy

The main ram Linear Variable Differential Transformers (LVDT) for the left rudder and the left stabilator actuators were replaced with optical transducers. The optical transducer is installed internal to the main ram cylinder and provides the same level of redundancy as the LVDT it replaces: dual for the rudder and quadruplex for the stabilator.

The Interface Converter Unit (ICU) interfaces the FCC's electrical inputs and outputs to the FACT optical signals. For the surface command (feedforward) path, the ICU converts the FCC surface command, an electrical current, to a digital optical signal that is transmitted to the Feedforward Remote Unit (FRU) located near the actuators in the rear of the aircraft. The ICU also receives a return optical signal from the FRU that represents the current flow through the actuator's electro-hydraulic valve. For the actuator position (feedback) path, the ICU provides the electro-optic interface to the optical sensors in the actuators; this interface is the Electro-Optic Architecture (EOA) module. The EOA contains the optical sensor light source and decodes the optic signal returned from the sensor. The ICU modulates this signal so the FCC receives an input of the same form as the production FBW LVDT feedback. The ICU checks for errors in the position feedback and the command paths and relays failure conditions to the FCC which shuts down the appropriate system. The ICU also provides transient-free 28 volt power to its internal power supply that also provides power to the FRU. Two dual channel ICUs interface with two dual channel FRUs to provide four channels of redundancy to match the two dual channel flight control computers.

The Feedforward Remote Unit (FRU) converts the feedforward optical command signal into a current command for the actuator's electro-hydraulic valve (EHV). Thus from the actuator's viewpoint, this signal appears to originate directly from the FCC. The FRU also measures the actual current flow through the EHV and converts this to an optical signal for return to the ICU. This return signal is used for error monitoring within the ICU.

The ICU and FRU also provide data to an in-flight or ground data acquisition system so data can be monitored and recorded.

Switches will be provided in the cockpit by NASA-Dryden so that the pilot retains complete control over whether or not the FACT hardware is actively controlling the aircraft (Figure 3). Since a portion of the production electrical servo-loop rudder and stabilator systems are removed and replaced by the FACT system, the F/A-18 FCC mechanical reversion modes serve as a fallback when the FACT system is switched off. The Left Rudder Control Augmentation

System (CAS) Off switch causes the left rudder to revert to a trail-damped mode; the right rudder remains in the CAS configuration. The PMECH Reversion switch causes the left stabilator to revert to its mechanical mode. The F/A-18 FCC production system will put the right stabilator in the mechanical mode when the left stabilator reverts. Reset switches will also be available so that the pilot can re-engage the FACT system following a reversion.

6.1.2 System Monitors

The FACT system monitors are designed to detect failures in both the feedforward and the feedback paths. Upon failure detection in a particular actuator channel, the ICU intentionally sends a fault to the FCC to trip a production system monitor and cause a reversion in that actuator channel. Figure 9 and Figure 10 show the combination of the FACT monitors and the production monitors associated with the rudder and stabilator servo loops.

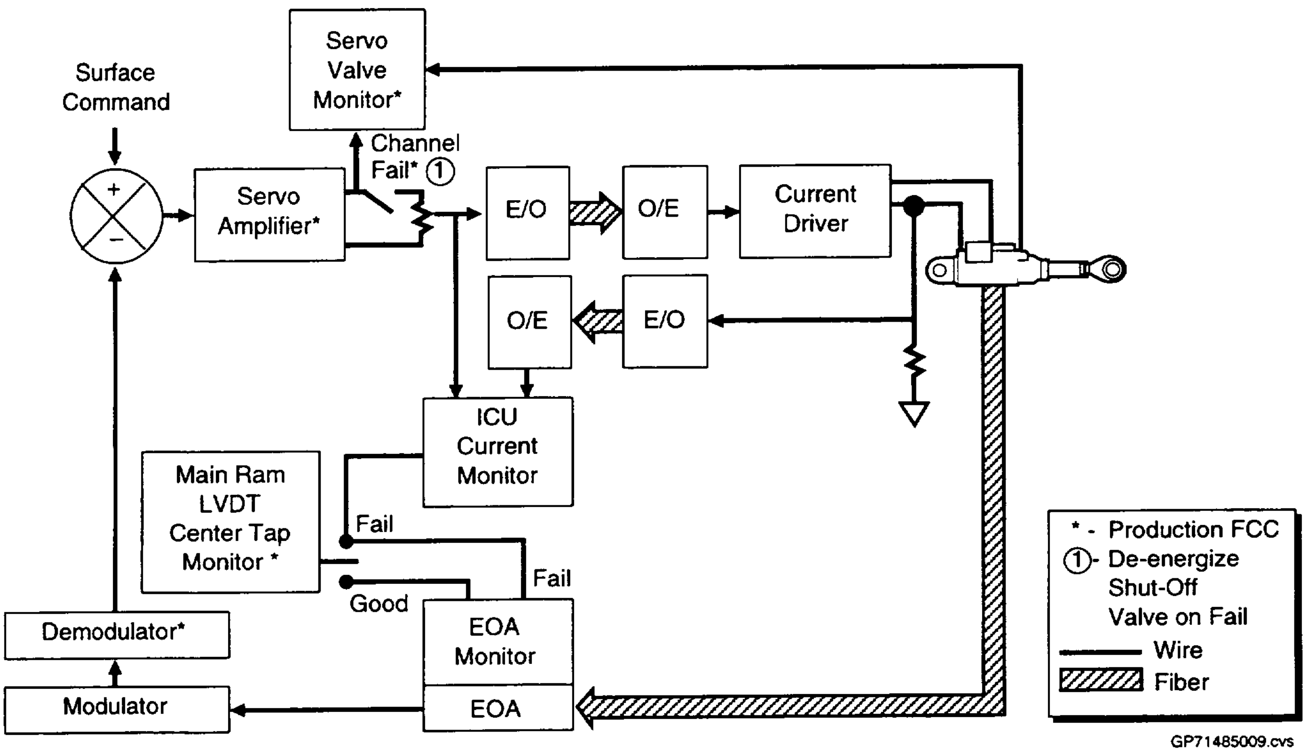


Figure 9. FACT Rudder Actuator Monitors

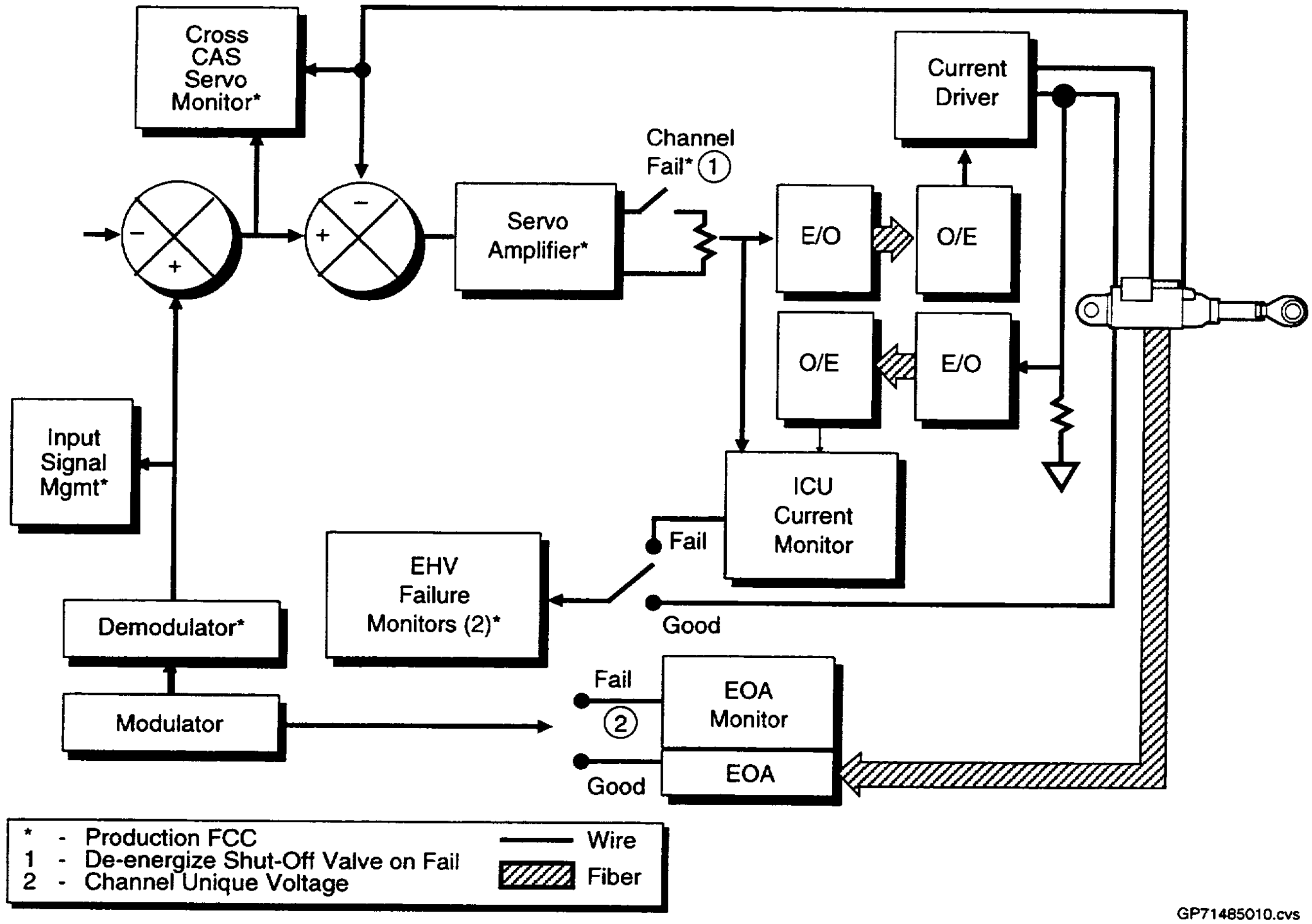


Figure 10. FACT Stabilator Actuator Monitors

The ICU Current Monitor compares the electrical feedforward command from the FCC with the measured current through the rudder EHV. A mismatch indicates a failure in the optical paths between ICU and FRU, or possibly a broken wire through the EHV. When tripped, the ICU Current Monitor creates a fault in the feedback signal to the FCC. In the case of the rudder, the LVDT Center Tap voltage is set to a fault level. The FCC production LVDT Center Tap Monitor then removes the servo-amplifier output and de-energizes the actuator shut-off valve for this channel. The left rudder reverts to a trail-damped state after failures occur in both FCC channels controlling the rudder. In the case of the stabilator, failure detection by the ICU Current Monitor causes the ICU to reset both EHV fault detection switch voltages to a fault level, which in turn causes the production EHV Failure Monitor to remove the servo-amplifier output and de-energize the actuator shut-off valve in this channel. (Note: two EHV fault detection switches are involved in each channel because the stabilator is driven by two independent hydraulic sources.) Stabilator reversion to the mechanical mode occurs after three FCC channels fail. In the mechanical mode, the stabilator actuators are controlled through mechanical linkages from the cockpit control stick.

The EOA Monitor is the primary fault detector for the FACT feedback path. When a fault is detected in the optical sensor or EOA hardware, the EOA module outputs a constant failure level voltage for the LVDT Center Tap. The FCC production LVDT Center Tap Monitor then detects the fail and prevents the FCC from using the failed control loop. In the case of the stabilator, the ICU sends a constant pre-defined LVDT voltage to the FCC. The voltage is unique to each of the four channels; thus, ambiguities associated with "two on two" failures are avoided. "Two on two" failures have occurred because the FCC chose the failed channels when two good channels had the same position and two failed channels had the same position. Because of the unique voltage assignments, the FCC Input Signal Management Monitor is able to correctly identify the failed channels which is an improvement to normal operation.

The monitor thresholds and reaction times were chosen so that the total time to shut down a channel following a failure by a FACT monitor is similar to the response time of a similar failure in the production electrical system. Latches are also provided in the FACT monitors to ensure that the production system monitor will shut down the actuator channel. Resetting the latches requires pilot action to prevent objectionable transients from automatically re-engaging the FACT system.

6.1.3 Pilot Vehicle Interface

The F/A-18 FCC cockpit status displays already provide sufficient information for the pilot to monitor the health of the FACT system. As shown in Figure 11, the pilot is advised of which channels are not engaged (i.e., the shut-off valve is de-energized) and when the actuator is in a reversion state. At the time of actuator reversion, the pilot also receives a voice alert. The Backup Caution Display provides redundancy should the display drivers fail.

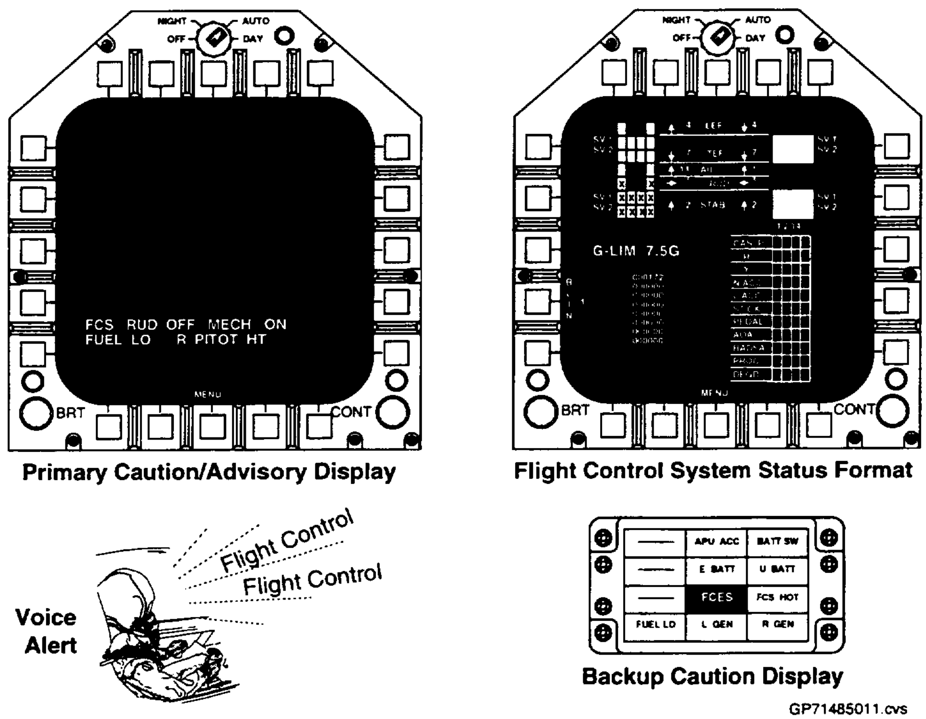


Figure 11. Flight Control System Pilot Information

The pilot will control when the FACT system is engaged. A switch will be installed in the NASA-Dryden SRA cockpit to control the left rudder. Turning the switch off will disrupt the rudder LVDT center tap feedback to the FCC in both rudder channels controlling the left rudder. The pilot will be able to control engagement of the stabilator surfaces through the Flight Test Control Panel (FTCP) in the cockpit. This panel, which was used in the development phase of the production F/A-18 flight control system, allows in-flight selection of the F/A-18 degraded flight modes through discrete commands to the FCC. By selecting mechanical reversion in the pitch axis, the pilot will be able to disengage the FBL and FBW commands to the left and right stabilators respectively through depression of the nose wheel steering button on the stick. Re engagement of FBL/FBW will be accomplished by depression of the paddle switch on the stick.

The production FCC Reset logic will be used to recover a channel following a failure and channel reversion. This would be done only after ground control personnel verify through instrumentation data that the FACT system has recovered to a valid operational state. All rudder and all feedforward loop failures can be reset without restriction, but stabilator feedback failures can only be reset for single channel failures that occur in no more than two of the four channels during a flight because of software latches in the production Input Signal Management logic. If the production software latches do not allow a channel to reset, the pitch axis resets to a degraded condition following an in flight FCC reset.

6.1.4 Aircraft Installation

The FACT equipment will be installed in the NASA F/A-18 (SRA) as indicated in Figure 12. The ICUs will be located in an avionics bay near the production FCCs. The vibration and temperature environments for the ICUs are relatively mild since the ICUs are in location built for avionics. The FRUs will be located in bays near the rudder and stabilator actuators. The environment of the FRUs is harsher than the ICUs, however, the environmental conditions still permitted using some commercial electronic components in the FRUs. The airworthiness environmental test plans are tailored according to the production environmental specifications for the ICU and FRU locations.

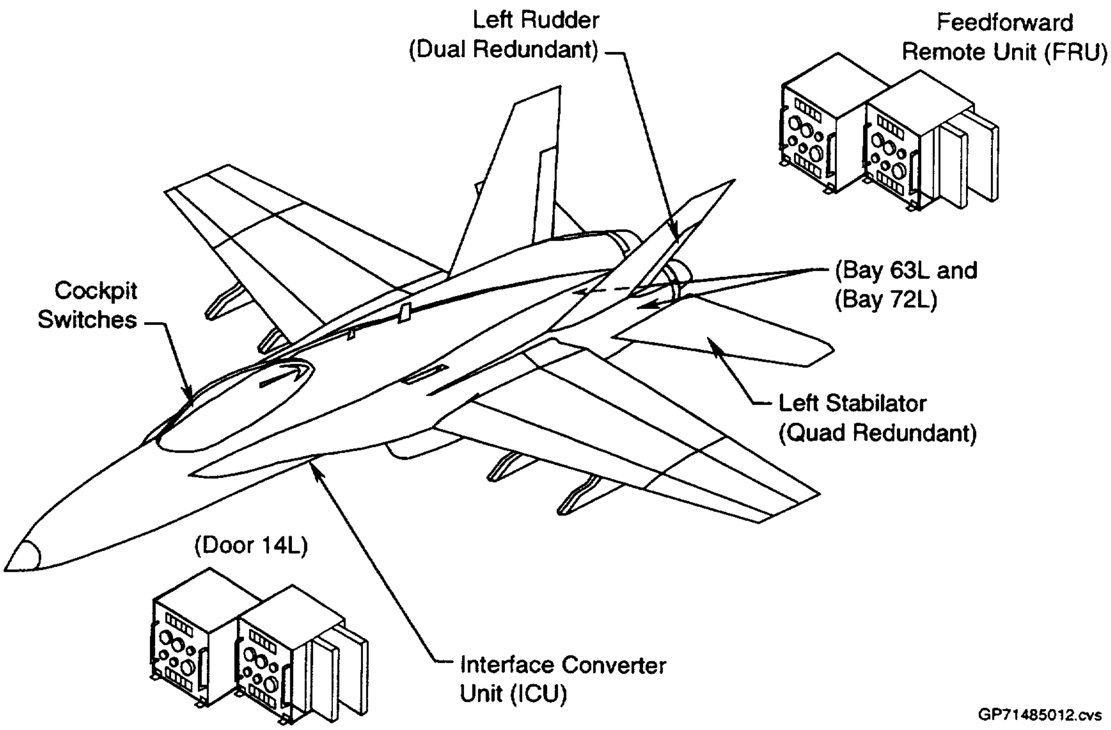


Figure 12. FACT Equipment Installation

6.2 Development

FACT system development covered a wide range of work in taking what was a well-defined concept and building a system ready for performance tests. Development tasks included designing and building new components, purchasing off-the-shelf or newly designed components, integrating components into the ICU and FRU chassis, creating a ground support station to monitor the FACT system, creating interconnecting cables for the system, initial testing of system components and the entire system to verify proper operation, and troubleshooting problems and fixing them. The following information covers the unique or significant items in system development.

6.2.1 Optic Brick Assembly for Sensors and EOA

The optic brick assembly is the main optic component of the rudder and stabilator optic sensors and EOAs. The optic brick is made of several pieces glued together to create wave division multiplexing (WDM) shown in Figure 13. A sensor optic brick receives light from a fiber, spreads the wavelengths of light over a code plate to align the wavelengths with a digital pattern, and gathers the wavelengths of light into a fiber. The EOA optic brick receives the wavelengths of light from the sensor and spreads those wavelengths over a charged coupled device (CCD) array. The CCD array converts the light into electric signals, and signal conditioning of the electrical signal reproduces the digital pattern of the code plate.

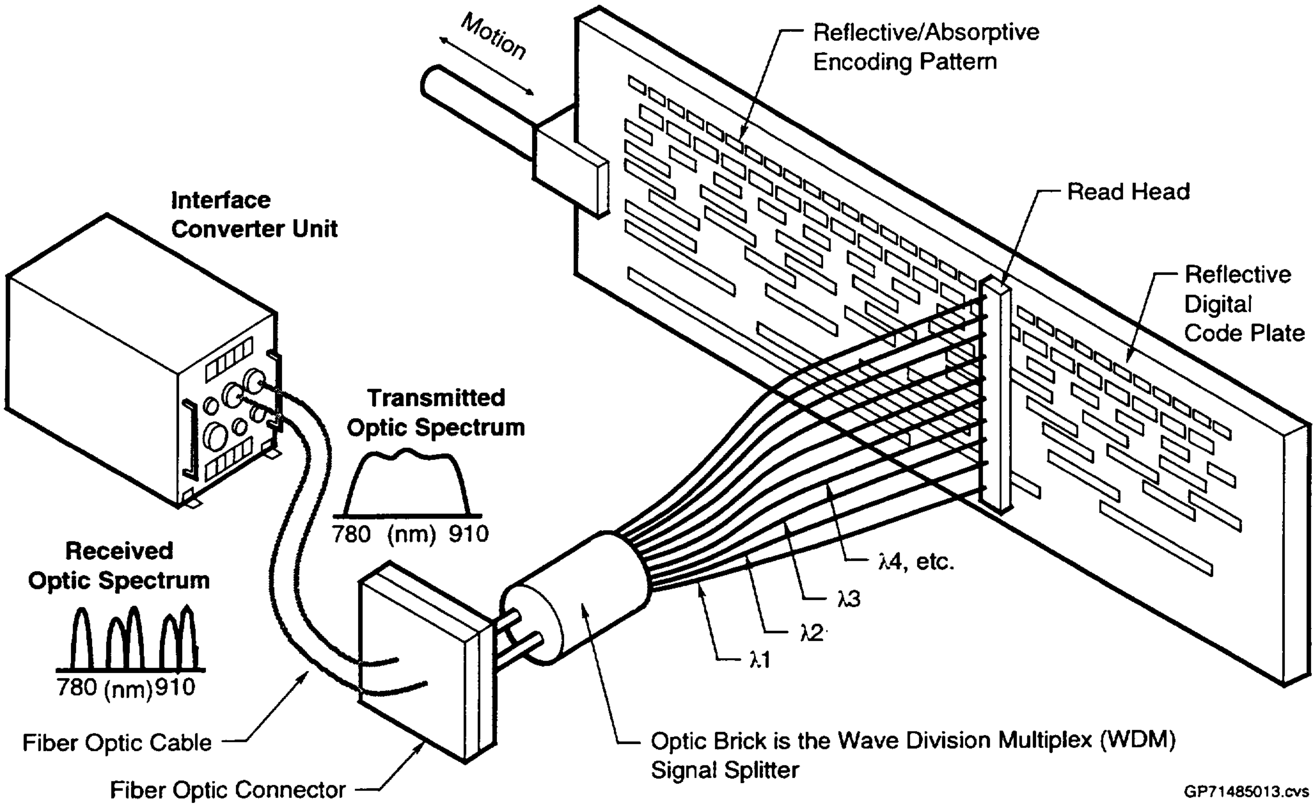


Figure 13. Optic Position Sensor and Wave Division Multiplexing

Problems assembling the optic brick pieces had serious consequences for the FACT program. Delays due to optic brick assembly and fitting the optic bricks into the stabilator cylinder caused the stabilator portion of the FACT program to be canceled. Also, many months of schedule slips resulted from delayed deliveries of EOAs, rudder sensors, and stabilator sensors.

Several problems were encountered in assembling the optic bricks. After several EOAs had been made, cracks appeared in the optic bricks. The adhesive, epo-tek 353, holding the optic brick pieces was blamed for stressing the glass during cure. Another adhesive, epo-tek 354, was chosen; it has less shrinkage during cure and a lower coefficient of expansion resulting in less movement of optic brick pieces at temperature extremes.

New problems occurred with the replacement adhesive. The adhesive needs precise amounts of ingredients so mixing techniques were developed to create proper mixtures. Even as the optic brick assembly process was improved and some good bricks were made, some optic bricks fell apart. Contamination was suspected, and the process was improved to eliminate contamination. Still optic bricks fell apart. Improvements were made in the way bricks were handled so more good bricks and the delicate fibers protruding from them survived handling. Investigations chased other potential problems and finally found humidity. Optic bricks were consistently and successfully made once humidity was controlled during cure.

The process is now good for producing optic bricks in small quantities. Unfortunately, almost a year passed before the assembly process was refined to consistently produce good optic bricks.

The design of the optic brick is an area for improvement. While the optic brick works well, it contains several parts that are difficult to assemble. Fewer pieces and easier assembly could make optic assemblies with more uniform performance and the possibility of mass production.

6.2.2 Digital Optical Sensors

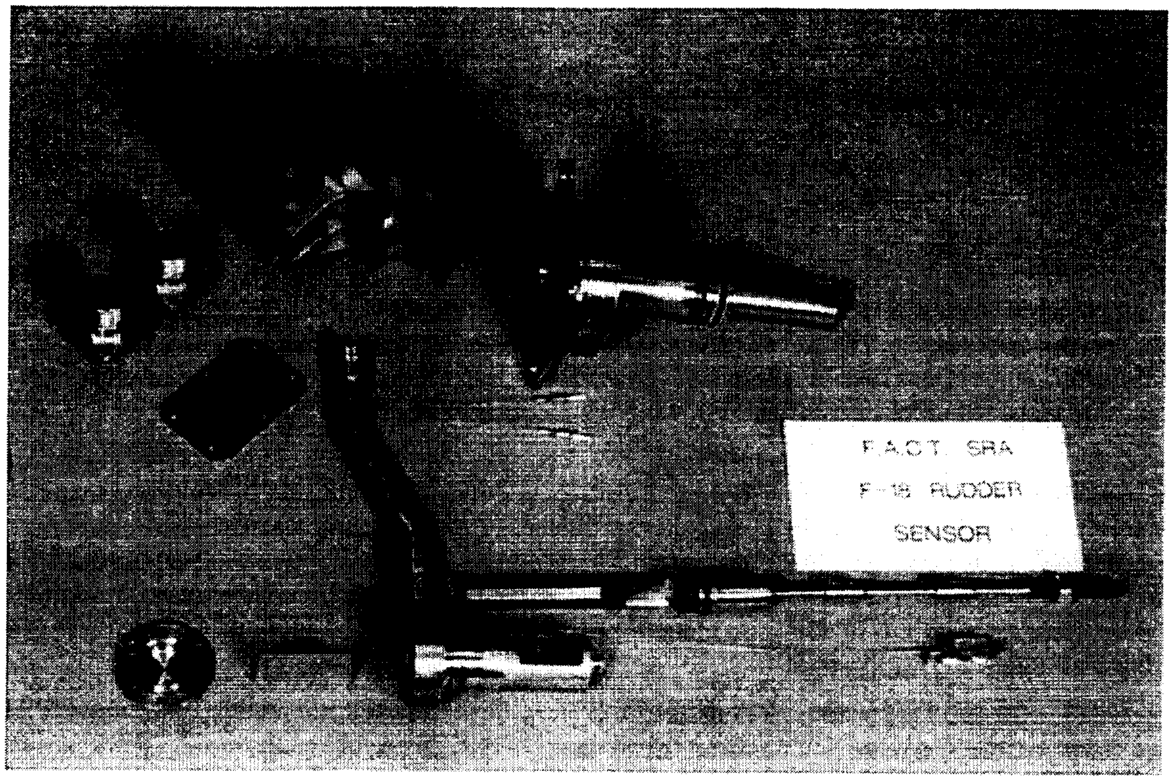
The optical sensors installed in the main ram cylinders use the optic brick to create wave division multiplexing and reflective digital code plates to create light patterns. The principle of operation is illustrated in Figure 13. The sensor receives light in the 750 to 900 nanometer wavelength range and diffracts and directs the light over the different tracks on the optical code plate. The reflective code plates alter the light intensity of sections of wavelengths to an on or off state. The reflected signals are recombined as a wavelength encoded digital pattern. Sensor redundancy is achieved through multiple read heads, multiple digital code patterns on a single metal plate, and input and output fiber pairs for each channel. Sensor measurement performance is summarized in Table 1.

TABLE 1. SENSOR PERFORMANCE

Sensor	Range	Resolution	Accuracy
Stabilator	±3.590 in.	0.0072 in.	0.0036 in.
Rudder	±0.715 in.	0.00140 in.	0.0007 in.

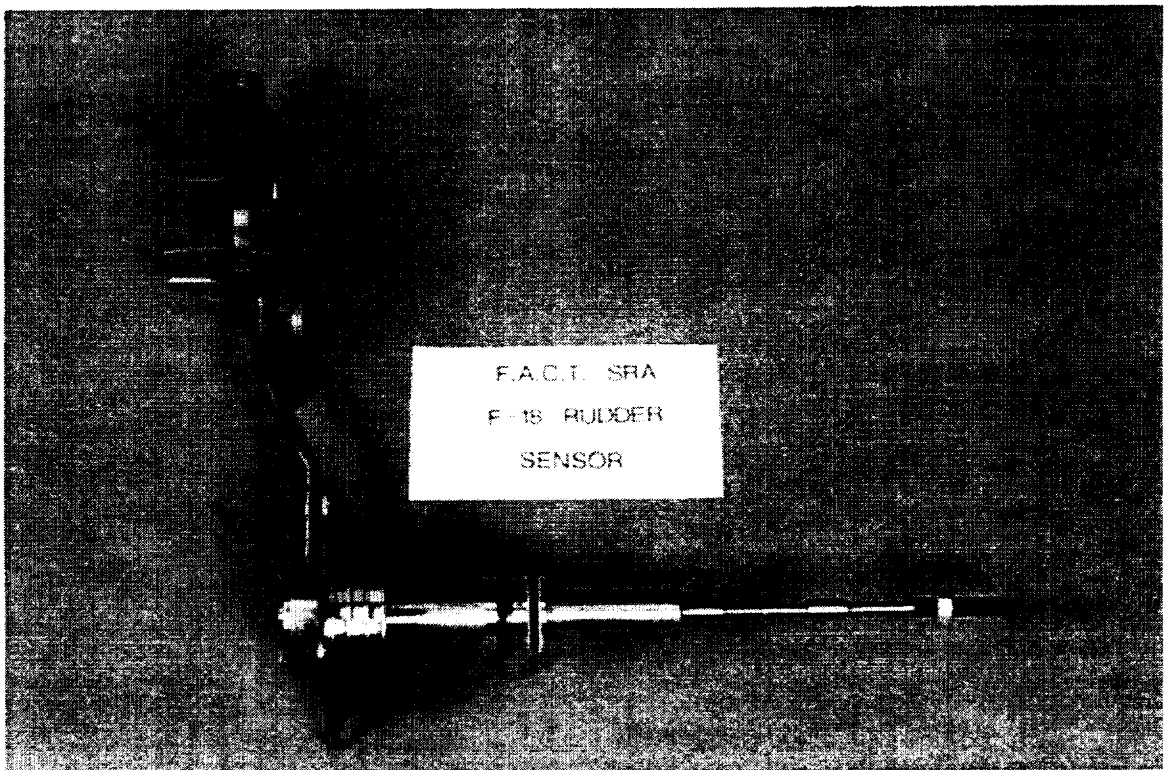
GP71485014.cvs

Figure 14 through Figure 16 show the rudder sensor from unassembled to installed in a rudder actuator. Figure 14 shows the unassembled pieces of the rudder sensor: the connectors and connector housing (upper left), the housing for the optic code plate and read head (upper right), optic termini and fiber attached to optic read head (middle and lower right), read head mounting and tube for fiber (middle and lower center), and code plate attached to shaft that attaches to the rudder actuator main ram cylinder (lower middle and right).



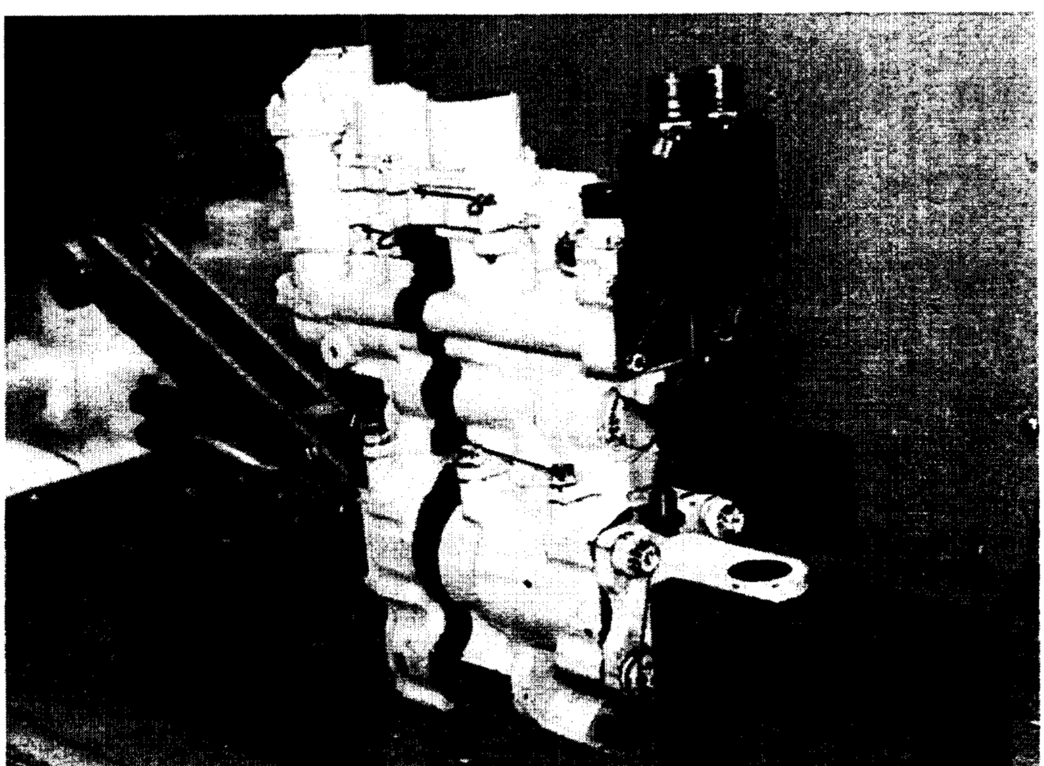
GP71485015.cvs

Figure 14. Unassembled Rudder Sensor



GP71485016.cvs

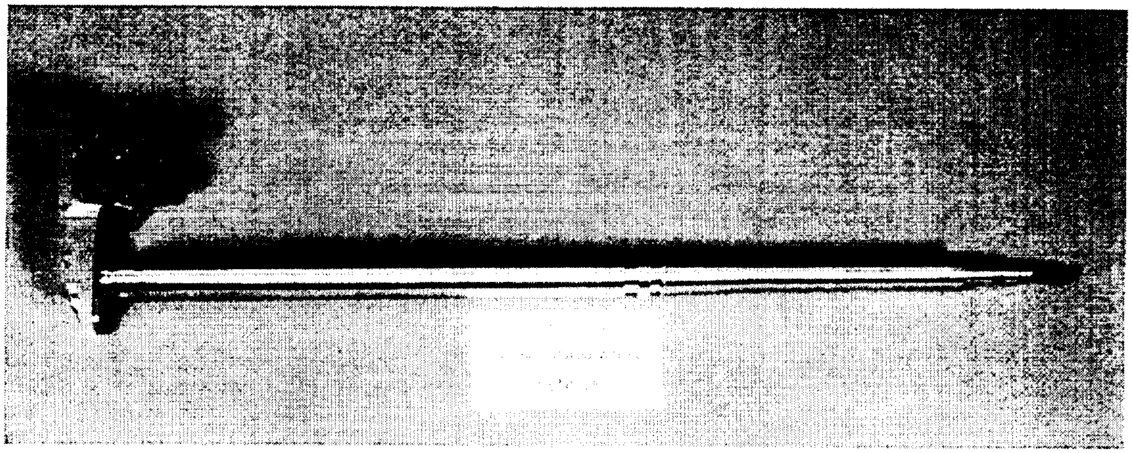
Figure 15. Assembled Rudder Sensor



GP71485017.cvs

Figure 16. Rudder Sensor Installed in Rudder Actuator

Figure 17 shows the assembled stabilator sensor. A stabilator sensor was not installed into a stabilator actuator due to delays in assembling and creating a working sensor.

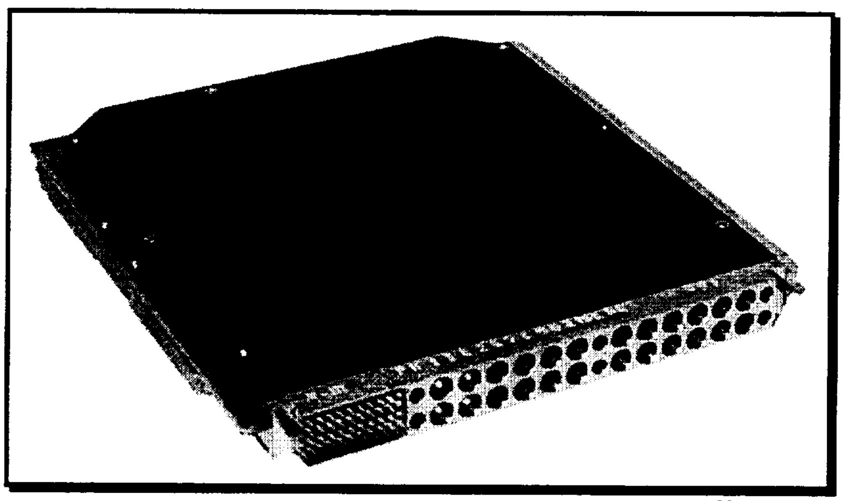


GP71485018.cvs

Figure 17. Assembled Stabilator Sensor

6.2.3 Electro-Optic Architecture Module (EOA)

The Electro-Optic Architecture (EOA) optic decoding module was designed to the FACT program functional requirements and the Standard Electronic Module (SEM) physical requirements of Naval Air Warfare Center (NAWC) Standard Hardware Acquisition and Reliability Program (SHARP).



GP71485019.cvs

Figure 18. EOA Module

The architecture of the EOA is shown in Figure 19. A light emitting diode (LED) provides light to an external passive sensor that creates a light signal. One LED is provided for each sensor so the amount of light to each sensor can be varied to reduce variations due to sensor attenuation. The light signal is diffracted and directed on one of two linear charge coupled device (CCD) arrays that convert the optic signal to electric. While one sensor signal is being received on one CCD array, another sensor signal on the other CCD array can be decoded. After signal conditioning, the digital signal processor (DSP) decodes the signal into the sensed position and performs extensive error monitoring on the signal. This self-monitoring is needed to notify the flight control computer of the validity of the actuator position feedback. Failure monitoring covers the loss of an LED, a broken fiber, invalid sensor signal, faulty CCD operation, or EOA processor hardware failure. When a failure is detected, an EOA fail discrete to the signal conditioning Interface module in the ICU causes a fail signal to be sent to the FCC so a reversion occurs in the FCC actuator channel.

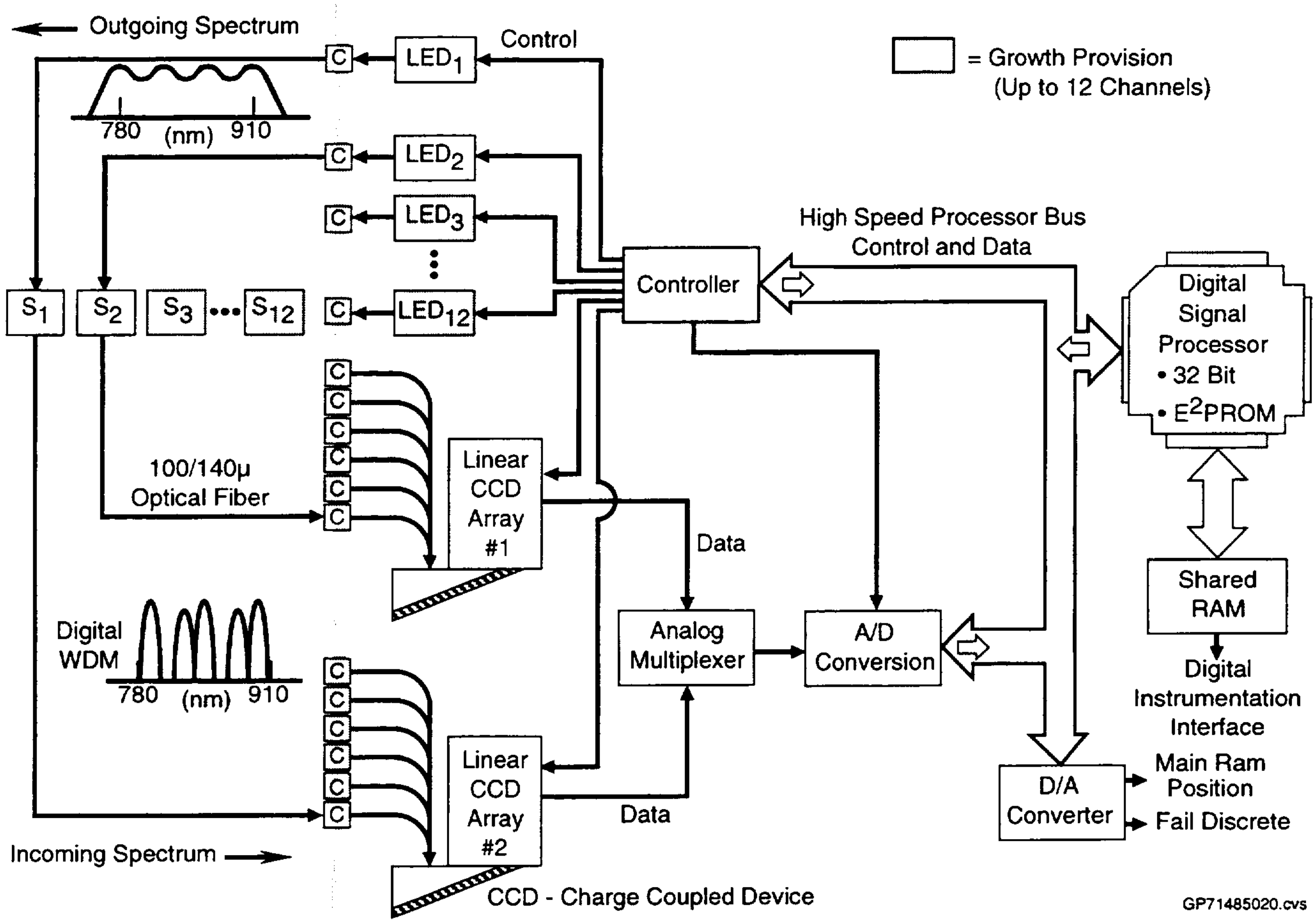


Figure 19. EOA Module Architecture

EOA error monitoring reports decode errors and decode failures. The FACT system can operate with some decode errors but no decode failures. The EOA tests the raw sensor signal to determine if the signal is good, and the EOA tests the decoded sensor position to determine if the position is valid. A failure in the raw signal or decoded position is a decode error. Fifteen consecutive decode errors are a decode failure that causes a reversion in the failed channel. To reset a decode failure, 2500 consecutive successful decodes are needed. At an example decode update rate of 500 hertz, a decode fail would last five seconds beyond the time the EOA stopped having decoding errors.

The EOA can accommodate up to twelve optical sensors, but the sensor signal update rate diminishes as the number of sensors increases due to the use of a single processor to decode all of the sensors.

In the FACT architecture, each EOA decodes up to two sensors. EOAs in two of the four flight control channels decode both the rudder and stabilator sensor. Both sensor positions are updated at a minimum rate of 400 Hertz with a maximum latency of 3.5 milliseconds. For the EOAs in the two flight control channels that process only the stabilator sensor, the update rate increases to a maximum of 830 Hertz while the latency decreases to a minimum of 1.5 milliseconds. The actual update rate and latency are a function of the time to decode, required CCD integration time that may vary for any decode cycle, and the number of sensors being decoded. The integration time for the stabilator or rudder optic sensor was limited to a maximum of two milliseconds in order to meet the minimum update rate of 400 Hertz.

The update rate and data latency of future EOA modules can be better than stated above. The EOA manufacturer, AlliedSignal, increased the update rate and decreased the data latency during the FACT program and had plans for even better performance that were not implemented.

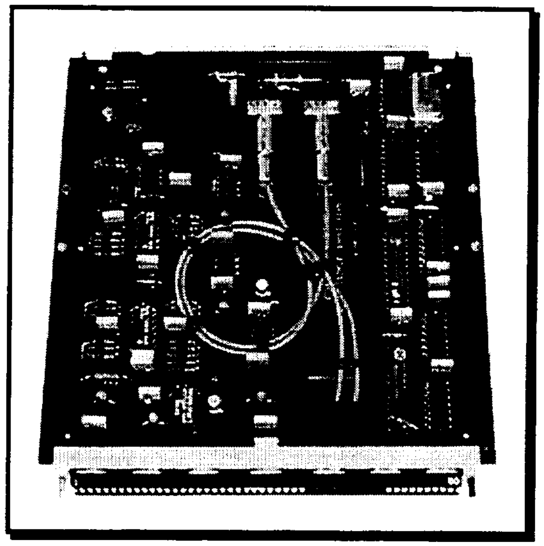
The EOA performance in decoding the rudder position is affected by the absence of the stabilator optic sensor due to the still active stabilator software in the EOA using the maximum integration time to try to decode the missing stabilator sensor. The EOA software to process the stabilator was left in the EOA to eliminate retesting the EOA, and no jumper was installed from the EOA optic source to EOA optic receiver for the stabilator sensor to eliminate the possibility of the jumper opening and changing the total EOA integration time and thus the update rate of the rudder sensor. All rudder testing was performed without the stabilator source to receiver jumper to create maximum delays in rudder position processing. The tests proved the system operates well with the delays in rudder position processing.

6.2.4 Feedforward Module

Two Feedforward modules work together to transmit a command signal from the FCC to the actuator. One module is in the ICU to interface with the FCC, and one module is in the FRU to interface with the actuator. The modules use the same method to transmit command information to each other over fiber. For transmitting, the electric analog current signal is converted to a voltage signal, converted to digital, encoded into a manchester signal, and converted to optic. For receiving, the optic signal is converted to electric, manchester decoded, and converted to an analog signal. The signal update rate from ICU to FRU is about 27 kilohertz which is well above the one kilohertz update rate needed for the F/A-18 fly-by wire system.

6.2.4.1 ICU Feedforward Module

The Feedforward module in the ICU provides the FACT optic interface to the electric FCC and provides error monitoring of the FACT command path. The ICU Feedforward module reads the command from the FCC, wraps that electric signal back to the FCC for electronic hardware stability and FCC error monitoring, converts the electric command signal to optic, and sends the optic command signal to the FRU.



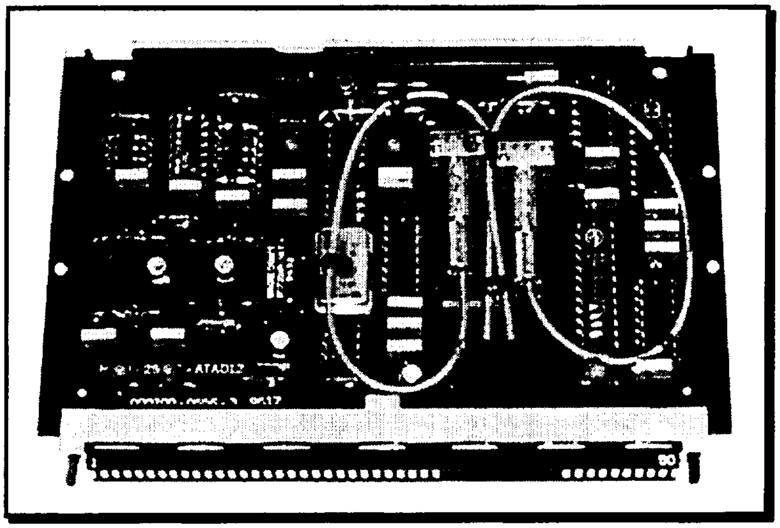
GP71485021.cvs

Figure 20. ICU Feedforward Module

The ICU Feedforward module also performs error monitoring functions. An optic signal representing the actual current through the actuator is received from the FRU Feedforward module for comparison to the FCC command. A mismatch error occurs if the FCC current command differs from the actuator current by 3.2 milliamperes which corresponds to the actual versus model current threshold in the flight control computer. The range of the FCC and actuator current is +/-8 milliamperes. When a mismatch occurs, several actions are taken. A relay opens interrupting the FCC command from the ICU Feedforward module, the transmitting LED is turned off to notify the FRU of the error, and a discrete is sent to the Interface module to declare an error. The discrete only lasts for five seconds which is long enough for the error to be presented to the FCC that shuts off the channel, but the relay remains open and the transmitting LED off until the ICU Feedforward module is reset.

6.2.4.2 FRU Feedforward Module

The Feedforward module in the FRU provides the FACT optic interface to the electrically controlled actuator and provides control loop closure around the actuator. The FRU Feedforward module reads the optic command from the ICU, converts that command to an electric signal, sends the electric command to the actuator, reads the actual current through the actuator, converts that current to an optic signal, and sends the optic signal to the ICU for error monitoring. Only Feedforward modules are in the FRU.



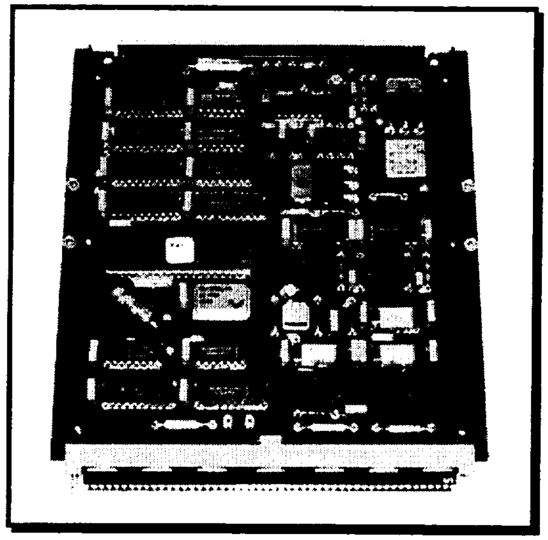
GP71485022.cvs

Figure 21. FRU Feedforward Module

The FRU Feedforward module performs error monitoring by detecting the validity of the optic manchester encoded signal from the ICU Feedforward module. If the signal is not valid, a relay opens interrupting the command current to the actuator. A broken fiber or greatly attenuated signal to the FRU can cause an invalid signal, or the ICU Feedforward module can cause an invalid signal if it detects an error.

6.2.5 Interface Module (IM)

The Interface module provides most of the ICU interfaces to the aircraft through three independent sections: input power switching, actuator position modulation, and instrumentation interface. (The Feedforward module directly interfaces with the FCC to obtain the command signal.) The power switching and modulation sections affect the actuator control loop and thus aircraft flight. The instrumentation section provides system status to an external data acquisition system for insight into system operation and does not affect aircraft flight.



GP71485023.cvs

Figure 22. Interface Module

6.2.5.1 IM Power Switching

The power switching function uses aircraft battery power to provide the power supply in the ICU with sufficient voltage, above sixteen volts, to operate normally for up to nine seconds after the aircraft main 28 volt power drops below 16.7 volts. The FCC has a similar input power switching function that supplies battery power to the FCC power supply for seven seconds. The FACT system will operate longer than the FCC so the FACT system will not affect the FCC if the aircraft main 28 volt power fails. After a switch to battery power and nine seconds, the switching circuit switches to aircraft main 28 volt power regardless of the voltage so battery power is removed from the FACT system power supply, and a relay opens preventing battery power to the switching circuit. The aircraft main 28 volt power must rise to 25 volts before the relay energizes and allows battery power to the switching circuit thus rearming the ability to switch to battery power. If the aircraft main 28 volt power rises above 18.6 volts before the battery has been on for nine seconds, the power supplied to the power supply switches back to aircraft main 28 volt power.

6.2.5.2 IM Signal Modulation

The signal modulation function converts the dc voltage analog position signal from the EOA to a Linear Differential Variable Transformer (LVDT) like signal for the FCC and provides switches to introduce errors so FCC error monitors can be tripped. The ac voltage excitation signal normally supplied to an LVDT is reduced in amplitude to the voltage needed for correct scaling of the analog position and multiplied with the EOA analog position to create the LVDT like position signal. Creating the LVDT like position signal is the same for rudder and stabilator actuator signals, however, introducing errors into the signals is different between the rudder and stabilator actuators due to the different ways the FCC monitors those signals.

The FCC uses a hardware centertap monitor for rudder position signal error monitoring and software comparisons of the four channels for stabilator position signal error monitoring. For the rudder, the difference between the high and low wires of the rudder position signal is the position; the sum of the high and low wires of the rudder position signal, with respect to a centertap, is a constant voltage. If that constant voltage changes, an error has occurred. The FCC stabilator error monitor compares each channel's signal to every other channel's signal. If a channel's signal differs from all other signals, that channel is not used to determine the stabilator position.

A constant rudder centertap signal from the EOA is used to create the centertap reference for the FCC rudder error monitor. The constant dc voltage from the EOA is multiplied with the same excitation used to create the rudder position signal. The resulting ac voltage is in phase with the position signal and is used as the center value for the high and low wires of the rudder signal. The sum of this signal is a constant voltage. When the EOA detects a fail, the EOA analog centertap value changes to zero instead of the normal 7.3 volts. When the Feedforward modules detect a fail, a discrete signal to the Interface module switches the constant dc voltage from the EOA to zero volts. In both cases, the FCC rudder centertap error monitor detects an out of range value and shuts off that channel within the same amount of time as the production system would shut off the channel.

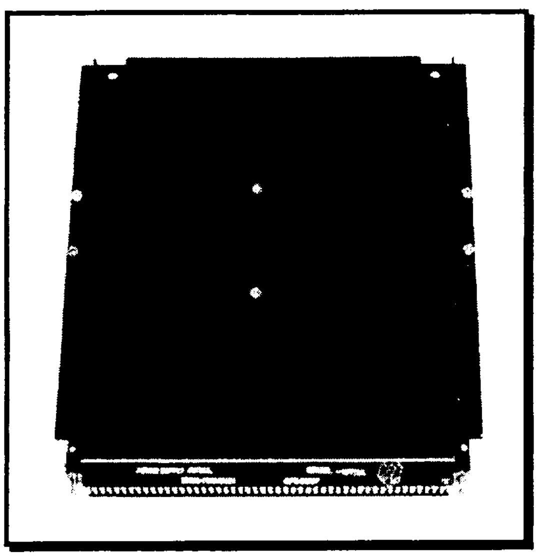
A stabilator signal failure is introduced by two methods depending if the fail is detected by the EOA or Feedforward modules. When the EOA detects a fail, the dc voltage analog position signal from the EOA changes to that channel's constant fail voltage, and the EOA sends a discrete that switches the dc voltage analog position signal from the EOA to the constant fail voltage. The constant fail voltage is obtained from the ICU backplane and is unique for each channel to avoid "two on two" failures. When the Feedforward modules detect a fail, the Feedforward module in the ICU sends a discrete that opens a relay that interrupts the stabilator differential pressure signals to the FCC. The differential pressure signals are the FCC error monitors for the Electrohydraulic Valves (EHV) that convert electric signals to hydraulic motion that moves the actuator ram. The FCC detects the fail signal and shuts off the channel as a result of the EOA or Feedforward module detecting a fail. The time to shut off the channel when the EOA detects a fail is the same amount of time as the production system. When the Feedforward modules detect a fail, the channel is shut off faster than a corresponding production system command failure because a different and faster FCC error monitor is tripped.

6.2.5.3 IM Instrumentation Interface

The instrumentation interface function gathers status from the FACT system and presents that status to an external data acquisition system. Status comes from the Power Supply, Interface module power switching function, Feedforward modules in the ICU and FRU, and EOA module. A microcontroller stores the latest data and delivers it when an external data acquisition system requests data. There is a serial link to the System Research Aircraft (SRA) data acquisition system and a parallel link to the PC ground support data acquisition system.

6.2.6 Power Supply Module

The Power Supply module located in the ICU provides power to the ICU and FRU.



GP71485024.cvs

Figure 23. Power Supply Module

The power supply modules were MIL-P-29590/8 supplied by NAWC-Indianapolis. The triple output switching supplies are 75% efficient taking 28 volts input and supplying ten amperes at five volts, one ampere at fifteen volts, and one ampere at negative fifteen volts. The switching frequency is 300 kilohertz. The supplies were designed to MIL-STD-704D requirements.

Electromagnetic compatibility tests performed by NAWC-Indianapolis showed the power supplies created conducted emissions beyond one megahertz that exceeded the MIL-STD-461C limits. An EMI filter was needed on the input power lines to attenuate the conducted emissions.

6.2.7 Electromagnetic Interference (EMI) Filter

An EMI filter was used to filter the aircraft main 28 volt power and battery power supplied to the ICU and to attenuate ICU power supply emissions back onto those power lines in order to meet MIL-STD-461C and MIL-STD-704 requirements. The filter consisted of off-the-shelf EMI filters and a custom aluminum chassis.

The off-the-shelf EMI filters were Spectrum Control, Inc. low pass filters, part number 51-353-100. The attenuation versus frequency curve of the filter is shown in Figure 24. The assumed filter source impedance of the 28 volt aircraft generator was 0.3 ohms, and the load impedance of the power supply was three ohms.

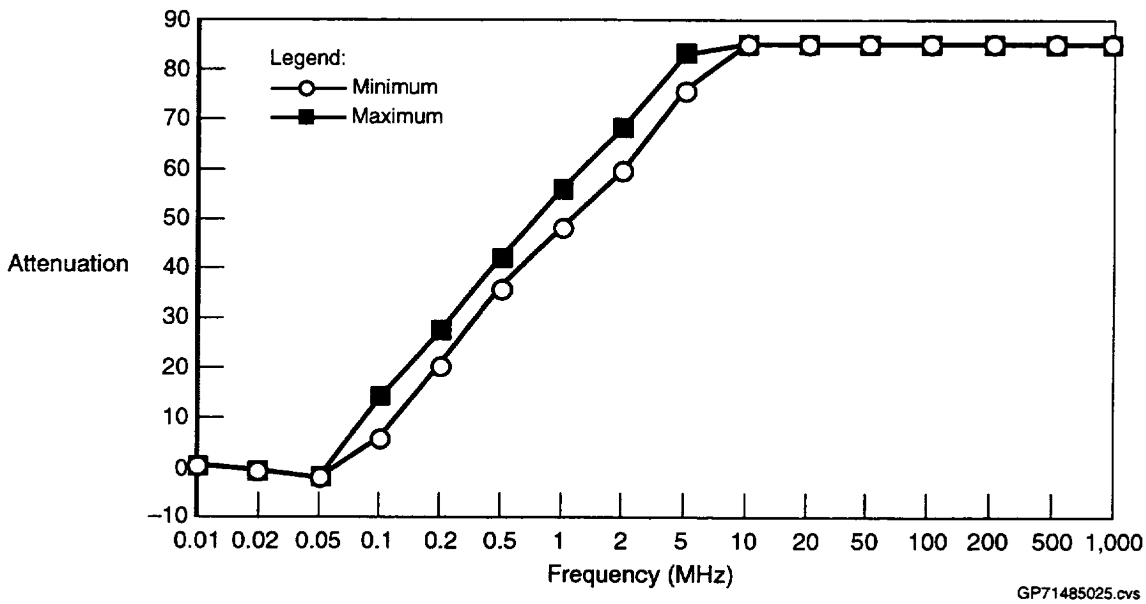


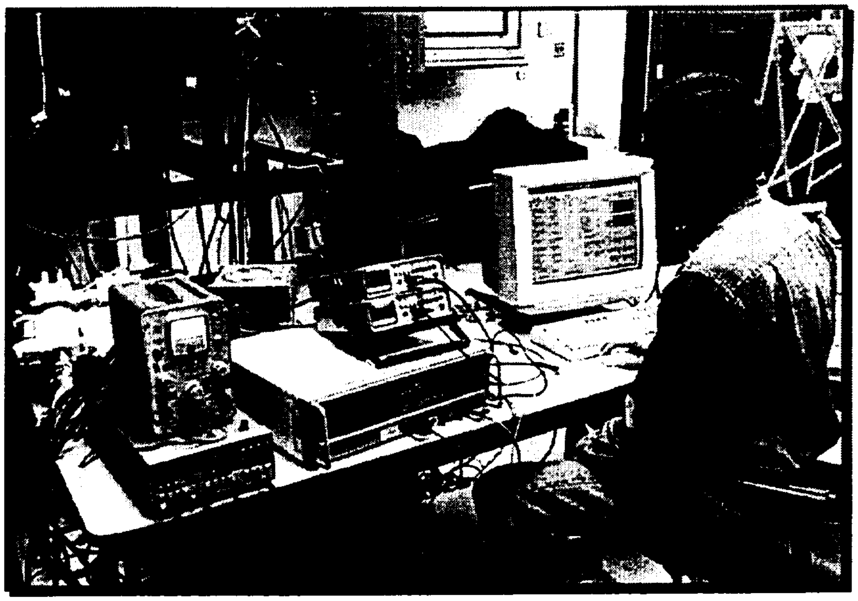
Figure 24. EMI Filter Attenuation vs. Frequency

The EMI filter chassis was a custom machined aluminum box made small enough to fit in the space between the ICU front panel with all the external connectors and the ICU motherboard. The filter was made with five sides and a cover plate that held the front panel power connector and attached to the inside of the ICU front panel. The off-the-shelf EMI filters were screwed into the filter chassis and wired to the connector. The cover plate attached to the filter chassis with screws and compressed an EMI gasket. The cover plate attached to the inside of the ICU front panel as an external connector and made metal to metal contact for the filter chassis electrical ground.

6.2.8 Ground Support Equipment

The ground support equipment was an IBM compatible personal computer (PC) based data acquisition system using National Instruments data acquisition boards and LabVIEW software. One analog acquisition board and one digital acquisition board were used to collect data from the ICU and FRU. Several user interface screens were created to easily view that data during component, system, and airworthiness tests.

The PC acquisition system worked well for monitoring and storing ICU and FRU data. The development time, including creating easy to read user interface screens, was relatively short for such a complex acquisition system, and making changes was easy. The PC acquisition system reduced the development and test time for the FACT system over using several pieces of stand alone test equipment, and it provided data storage.



GP71485026.cvs

Figure 25. Ground Support PC and Other Test Equipment (Temperature/Altitude Testing)

The PC acquisition system did take more time to maintain than was expected due to problems that were difficult to troubleshoot because it was based in a PC. A PC crash corrupted the host software just enough to cause slight acquisition system problems. The data acquisition boards touched the PC frame and suffered damage two times before the cause of the problem was found. There was also an electromagnetic interference (EMI) susceptibility problem that caused the data acquisition system to lose track of the correct data order. The EMI problem occurred with external power switching transients. Troubleshooting and fixing these problems took time away from developing and testing the FACT system.

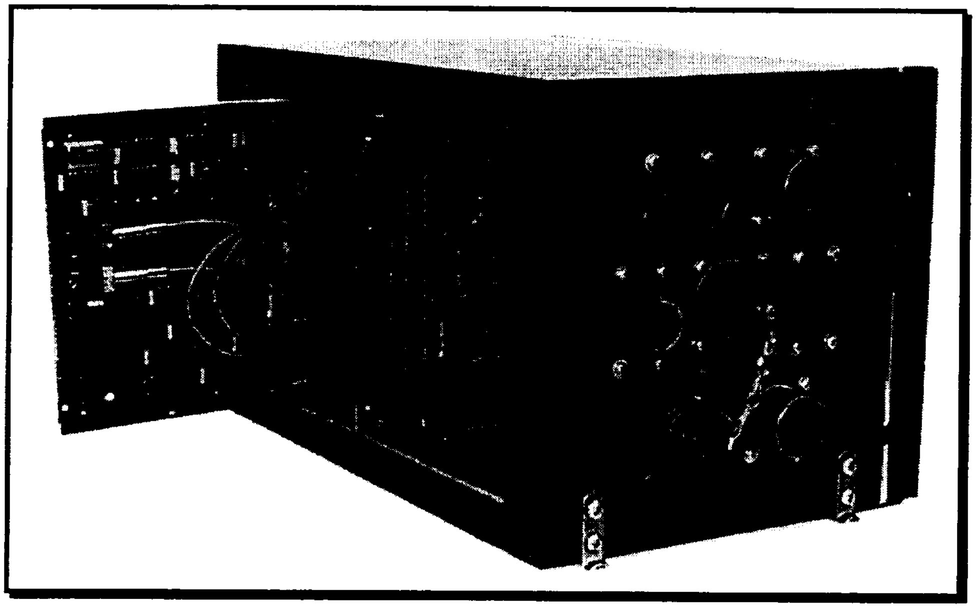
6.2.9 Optic Connector for Module to Backplane

A module to backplane fiber optic blind mate connector was developed for the FACT program. When the electro-optic modules were being designed, there was not a connector for fiber optic termini that were affordable and easily terminated. A connector existed for the G&H Technology Incorporated expanded beam termini that were many hundred dollars per mated pair and were hard to terminate. Newer high density connectors like AT&T's ROC were still in development and not assured to be complete in time for FACT. FACT team members decided to develop a connector to hold the MIL-T-29504 /4, /5, /10, and /11 termini. The /4 and /5 termini are for MIL C 38999 connectors, and the /10 and /11 termini are for MIL C 83723 connectors. Those termini are used extensively in flight applications, easy to terminate by experienced people, and less than one hundred dollars per mated pair.

The connectors to hold the optic termini were designed as follows. For the feedforward electro optic modules, the longer /11 optic sockets were placed in the modules and the shorter /10 optic pins in the backplane to reduce the space needed behind the backplane. The EOA module was built with shorter /4 optic pins in the module so longer /11 optic sockets had to be placed in the backplane. The /4 and /11 optic termini are not normally mated but were able to be used together since they are from the same manufacturer, ITT Cannon, and the end face cross sections are identical. The connector was machined aluminum and black anodized. The mating halves were fixed with respect to each other and the backplane. Each optic terminus is held in place by a spring clip and allowed to float in the connector. The connectors mated easily and were forgiving of tolerance errors in the backplane holes for the connector. The optic connectors worked without problems during all tests including component, system, and environmental airworthiness tests.

6.2.10 Interface Converter Unit Chassis

The Interface Converter Unit (ICU) chassis was a 3/4 ATR composite chassis made by Courtaulds Aerospace and supplied by NAWC-Indianapolis. The aluminum rack inside the composite frame and covers held Standard Electronic Modules of size E (SEM-E) on a 0.6 inch spacing. A 0.75 inch aluminum spacer was added between the chassis and the backplane access cover to provide bend radius space for the fiber cables attaching the backplane to the front panel external connectors. The aluminum spacer was attached with added screws and electrically conductive silver epoxy to maintain EMI shielding. The composite chassis provided EMI shielding through electrically conductive EMI gaskets and close screw spacing on the covers. Compared to an equivalent aluminum chassis, the composite chassis is 40% lighter and equal in heat dissipation.



GP71485027.cvs

Figure 26. Interface Converter Unit

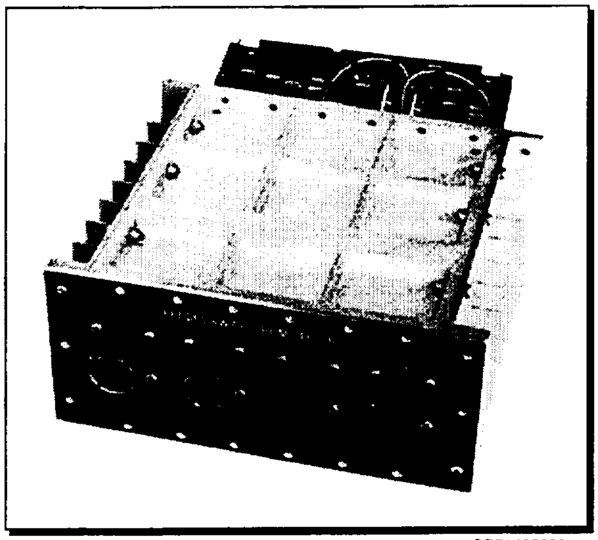
Each chassis was made to hold two FCC channels worth of modules, and two chassis were needed per aircraft. After the stabilator portion of FACT was dropped, only two FCC channels were needed since the rudders are dual redundant so each chassis held four modules: power supply, EOA, Interface, and rudder Feedforward modules.

The ICU had seven MIL-STD-38999 connectors on the front panel that supported two FCC channels. Each channel had one connector for fiber optic signals, one connector for electrical signals, and one connector for power. One connector was used for ground support equipment connections for both channels. Each connector, except for the power connectors in the EMI filter housing, had an EMI gasket between the connector and the inside front panel to aid in EMI shielding. The connector for fiber optic signals was a 38999 series III with MIL-T-29504/4 and /5 optic termini.

The two ICU chassis are to be located in Door 13L, an environmentally conditioned bay for avionics. A shelf was modified to hold the ICUs. The ICUs do not need vibration shock mounts or forced air cooling.

6.2.11 Feedforward Remote Unit Chassis

The Feedforward Remote Unit (FRU) chassis was a custom aluminum chassis provided by NAWC-Indianapolis. The chassis needed to be small to fit in odd spaces available in the rear of the aircraft near the rudder and stabilator actuators. The chassis held SEM-C size Feedforward modules that delivered the commanded position to the actuators. The card guide was machined into the chassis sides, the backplane was held in place by wedgeclamps, and the screws holding the sides together were close together to maintain metal to metal contact for EMI shielding.



GP71485028.cvs

Figure 27. Feedforward Remote Unit

The FRU had four MIL-STD-38999 connectors on the front panel supporting two FCC channels. Each channel had one connector for fiber optic signals and one connector for electrical signals and power. Each connector had an EMI gasket between the connector and the inside front panel to aid in EMI shielding. The connector for fiber optic signals was a 38999 series III with MIL-T-29504/4 and /5 optic termini.

Custom holding fixtures were made to hold the FRU chassis in the odd spaces available at the rear of the aircraft. Boeing designed the fixture for the Bay 72L location, and NASA-Dryden modified an existing fixture for the Bay 63L location. Environmental conditioning was not available at these locations, but the FRUs did not need vibration shock mounts or forced air cooling.

6.2.12 Relationship of Components within the System

Figure 28 shows the relationship of the components in the system. Section 6.1.1, System Overview, describes the FACT system functions. The component descriptions in section 6.2, **Development**, describe each component.

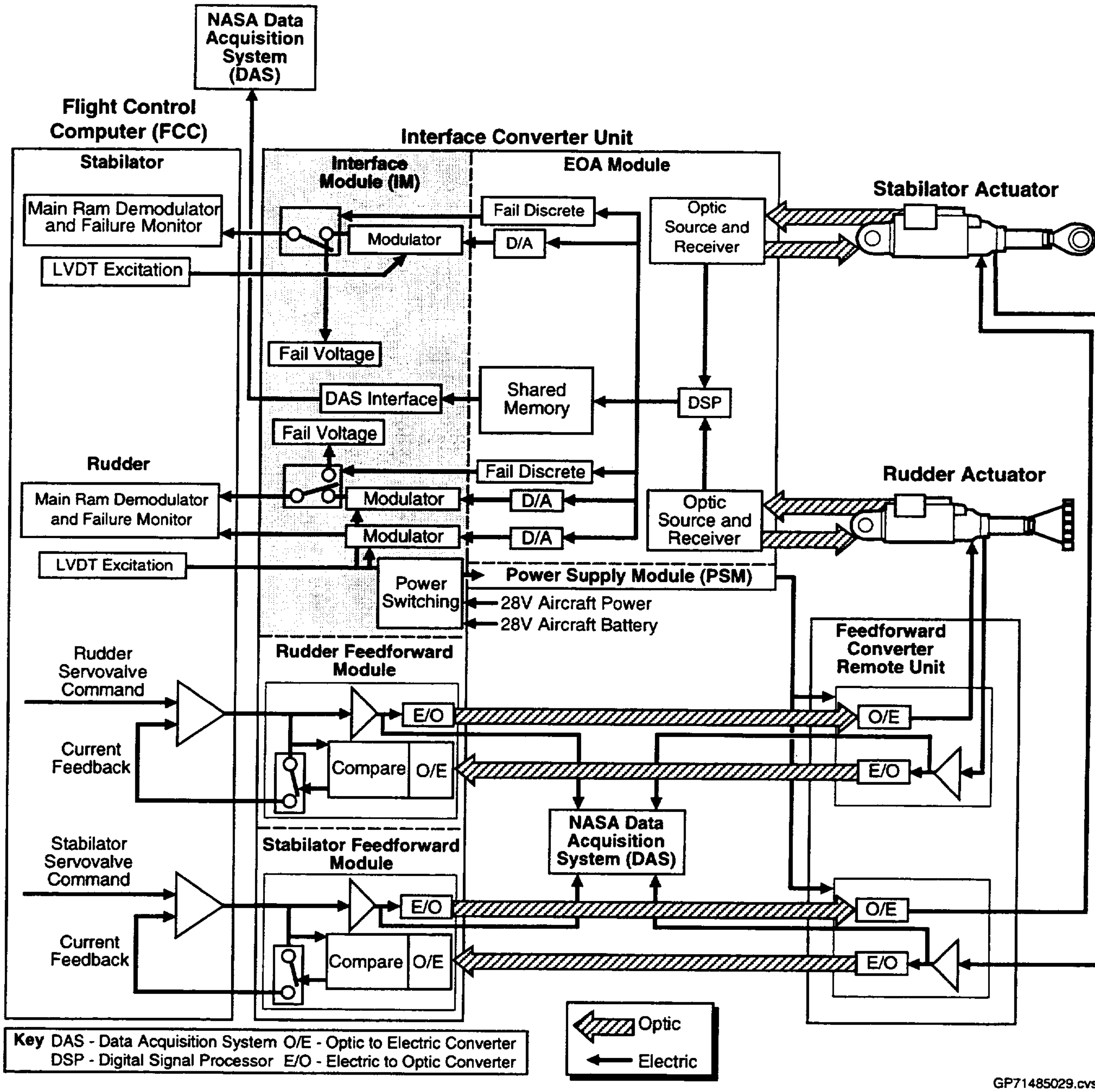


Figure 28. FACT Architecture (One Channel of Four)

7. FACT TESTS PERFORMED, TEST RESULTS, AND DISCUSSION

This section describes the FACT system with only the rudder actuator since the stabilator sensors were not completed in time to install in a stabilator actuator and test with the ICU and FRU.

The FACT equipment integrity and performance was verified for flight test through environmental stress screening to eliminate bad components; component tests to verify ICU and FRU functions; system tests to verify combined FCC, ICU, FRU, and actuator system performance and error handling through the FCC; and environmental airworthiness tests to verify the ICU and FRU can withstand the fighter aircraft environment. The tests were performed at Boeing St. Louis unless stated otherwise.

7.1 Optic Sensor Tests

AlliedSignal performed several tests to verify optic sensor performance and ruggedness. Each sensor was environmentally screened in vibration and temperature tests to provided confidence in the construction of each sensor. In a lifetime wear test, one optic sensor was subjected to 278,000 full stroke cycles and over 7,400,000 dither cycles with a 0.062 inch stroke. The sensor was fully functional after the test. Airworthiness tests of temperature, altitude, and pressure impulse tests were also performed on one optic sensor to validate the ruggedness of the design. The pressure impulse test was performed on a sensor in a rudder cylinder housing by Dowty Aerospace to verify the sensor could withstand stress from hydraulic pressure cycling. A performance test was used to check the sensor during the last cycles of airworthiness tests and after the environmental screening tests. The performance test checked linearity and accuracy by comparing the sensor position to a high precision reference. These tests readied the sensors for installation into actuators.

7.2 Actuator and Optic Sensor Acceptance and Airworthiness Tests

After a sensor was installed in a rudder actuator, Dowty Aerospace performed slightly modified acceptance test procedure for production actuators. The tests verified the performance of the actuator by testing proof pressure, seal leakage, friction, null position, and sensor output. Table 2 summarizes the FACT rudder acceptance tests in comparison with the F/A-18 production actuator acceptance tests.

One rudder actuator (or sensor for sensor only tests) went through airworthiness tests to verify the flight ruggedness of the FACT rudder actuator with the optic sensor. The vibration profiles were the vibration environments defined for the rudder actuator in the F/A-18. The airworthiness tests are summarized in Table 3 in comparison with the F/A-18 production actuator flight validation tests.

TABLE 2. RUDDER ACTUATOR ACCEPTANCE TEST PLAN

Acceptance Test	F/A-18 Acceptance Test Plan Description	FACT Acceptance Test Plan
Physical Defect Inspection	Parts subjected to structural or pressure loading shall be magnetic particle or penetrant inspected per MDC spec.	AlliedSignal performs inspection.
Examination	Inspect for dimensional requirements, nameplate, workmanship. Check electrical connections.	Extend for FACT to check optic connections.
Insulation Resistance		Actuator not affected by FACT.
Dielectric Test		Actuator not affected by FACT.
Proof Pressure	a) 4500 psi supply with return open b) 3000 psi supply and return c) 2 min. duration each at 120° $\pm$ 40° F with no leakage	Part of FACT actuator ATP.
Operation	5 full cycles without chatter or instability with both channels or one channel operating	Part of FACT actuator ATP.
External Leakage	250 cycles of $\pm$ 25% stroke about null at zero load with no chatter or instability. Rod seal leakage not to exceed 1 drop per seal.	Part of FACT actuator ATP.
Internal Leakage	With 3000 supply pressure and main ram static, measure leakage at return port.	For FACT, leakage at return port should be zero. Test leakage at the vent to atmosphere.
Output Travel	Main ram stroke is 1.43 inches.	FACT sensor measures full ram travel.
Main Ram Velocity		Actuator not affected by FACT
Overload Relief Operation		Actuator not affected by FACT
Damper Test		Actuator not affected by FACT
Main Ram Transducer Performance	Accuracy, tracking, null setting and phasing requirements must be tested installed in actuator.	Part of FACT actuator ATP.
Failure Transients Shutdown Time		Actuator not affected by FACT
Threshold	Largest sinusoidal input amplitude to servo at 0.1 Hertz without main ram motion should not exceed 0.05% ram stroke (.0725 ins.).	Part of FACT actuator ATP.
EHV Null Bias	Input current required to hold main ram static should not exceed 0.25 ma differential between two coils.	Part of FACT actuator ATP.
Frequency Response	Satisfy defined boundaries for gain and phase responses from 0.1 to 30 Hertz with input command peak-to-peak amplitudes of 1% and 7% full main ram stroke.	Part of FACT actuator ATP.
EHV Servo LVDT		Actuator not affected by FACT
Solenoid Valve		Actuator not affected by FACT
Pressure Switch		Actuator not affected by FACT
Pressure Port Check Valve Test		Actuator not affected by FACT
Production Duty Cycle	Applies to production units only.	Not applicable to FACT

TABLE 3. RUDDER ACTUATOR AIRWORTHINESS TEST PLAN (PART 1)

Preflight Test	F/A-18 Pre-Flight Validation Description	FACT Rudder Actuator Airworthiness Test Plan (Some test performed only on optic sensor)
Acceptance	Sequence of tests to be applied to each development unit	Conduct tests that exercise optical sensor or current command interface. See Rudder Actuator Acceptance Test summary table.
Life Cycling	Spectrum of load-stroke cycles prorated for duration of flight test with no evidence of excessive wear or leakage. For 300-hour flight test: • 5,000 full stroke, full load cycles; • 25,000 half stroke, half load cycles; • 70,000 10% stroke, 10% load cycles; and • 400,000 2% stoke cycles superimposed proportionately on above.	Allied performed wear test on sensor mask: • 278,000 full stroke cycles • 7.4+ million $\pm$2% stroke dither cycles. Wear test results apply to both rudder and stabilator designs and satisfies intent of a life cycle test for the sensor.
Piston Bottoming	Distribute 6,500 bottoming cycles against the full extend and full retract stops each into life cycle test cycles.	No tests planned. Structural integrity of actuators is not impacted by FACT.
Pressure Impulse Cycling	Pressure cycles from 1000 psi to 4050 psi to 1000 psi, with 10 msec minimum dwell at max pressure. For production unit, 1,000,000 cycles in extend mode and 1,000,000 units in retract. Leakage not to exceed External and Internal Leakage requirements of the Acceptance Tests.	Flight qualification sensor housing to be tested at actuator house for 300 hour flight test program (5% of production unit life) • 50,000 cycles with mask extended; and • 50,000 cycles with mask retracted. Check for leakage.
Temperature	• <u>High Temperature</u> 2 hour soak at 275^{0}F fluid temperature. Pressurize unit, operate for five cycles and check for leakage. Energize unit to test performance (threshold, null bias and frequency response), insulation resistance and dielectric. • <u>Low Temperature</u> 4 hour soak (3 hours for stabilator) at -40^{0}F. Pressurize unit, operate for five cycles and check for leakage. Energize unit to test performance (threshold, null bias and frequency response), insulation resistance and dielectric. • <u>Temperature Shock</u> Fluid temperature stabilized at -40^{0}F. Ramp chamber temperature from -40^{0}F to 240^{0}F, fluid from -40^{0}F to 275^{0}F within 200 seconds. Hold 1 minute. Stabilize chamber at 220^{0}F and hold 10 minutes. Three such cycles with full ram motion checks ($\pm$ 50% motion for stabilator) at 36^{0}F intervals or less.	Sensor tested separately: • 4 hour soak at -65^{0}F • Ramp to 275^{0}F within one hour and soak for 4 hours. Allied can ramp within 30 minutes. • Ramp to -65^{0}F within one hour. Perform 10 such cycles. Check sensor static performance on two channels throughout test. Check sensor dynamic performance after test. The sensor shock test has been removed since the intent of this test is not related to the performance of the sensor. AlliedSignal to conduct test.

TABLE 3. RUDDER ACTUATOR AIRWORTHINESS TEST PLAN (PART 2)

Preflight Test	F/A-18 Pre-Flight Validation Description	FACT Rudder Actuator Airworthiness Test Plan (Some test performed only on optic sensor)
Temperature/ Altitude	Note: Altitude tests are NOT specifically referenced under Preflight Verification, Paragraph 4.1.6. Reference multi-step procedure in MIL-STD-810. Steps include following: • Altitude to 70,000 feet • Low temperature of -40^{0}F • High temperature of 160^{0}F • Perform 10 cycles Test performance.	Sensor is tested separately at ambient temperature: • 4 hour soak at SL pressure • Ramp to 50,000 feet pressure within 1 hour • 4 hour soak at 50,000 feet • Ramp to SL pressure within 1 hour Perform 10 such cycles. Check sensor static performance on two channels throughout test. Check dynamic performance after test. AlliedSignal to conduct test.
Vibration	Ground rules for sinusoidal vibration testing: • Total cycling/endurance time cover life of unit (6,000 hours for actuators). Rule of thumb for flight test is to scale times by 25% and to hold levels the same. • Resonance is defined as transmissibility > 2. • Intent of resonance dwell time is to establish performance compliance at resonance and structural integrity of unit. Ground rules of random vibration testing: • Endurance levels and duration cover life of unit (6,000 hours for actuators). Rule of thumb for flight test is to scale times by 25% and to hold levels the same. • Testing at "Performance Levels" and not "Endurance Levels" covers a 50-hour life. Set ram for surface neutral position.	<u>General Comments</u> • Structural integrity of actuator is NOT an issue. • Integrity and performance of sensor are issues. Tests performed with sensor installed in actuator provided by Navy. The vibration tests are performed at different ram positions due to potential differences in vibration response: surface neutral, fully extended, fully retracted and mid-stroke when different from surface neutral. During all vibration tests, the sensor output does not deviate by more than $\pm$ 1% or 1 least significant bit, which ever is greater, from the its value at the start of the test. During all vibration tests, all sensor channels are monitored using FACT ICUs, or functional equivalents, as required.
	<u>Resonance Survey</u> – 5 to 2000 Hz at $\pm$0.024 inch amplitude or $\pm$2g, whichever is less. Note resonance points. A "resonance" is a measured response with transmissibility of 2 or more relative to the driver	<u>Resonance Survey</u> – Frequency sweep from 5 to 2000 Hz at lesser of 0.024 inch double amplitude or $\pm$ 2g. Monitor two channels during test. Check accuracy on all channels after test. Surveys are performed for each ram position defined above.

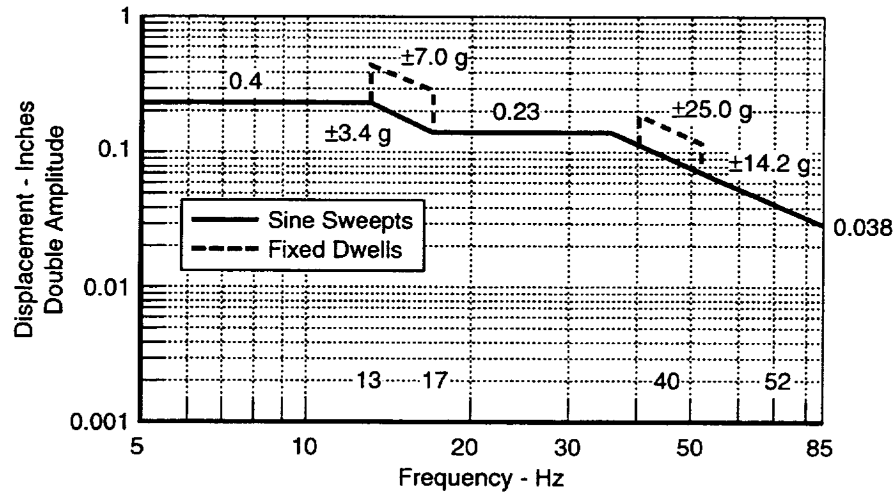
TABLE 3. RUDDER ACTUATOR AIRWORTHINESS TEST PLAN (PART 3)

Preflight Test	F/A-18 Pre-Flight Validation Description	FACT Rudder Actuator Airworthiness Test Plan (Some test performed only on optic sensor)
Vibration – continued	Rudder Sinusoidal Vibration Based on Region 5A. Cyclic sweeps from 5 to 85 Hz to 5 Hz with 10 minutes per sweep and with dwells at two most severe resonances other than 15 and 46 Hertz. Total test time is 90 minutes each axis, including 30 minute dwells at resonances. 173 minutes/± 7g and 107 minutes/±25g dwell apply to 15 and 46 Hz are conducted in lateral axis, respectively. Performance compliance must be demonstrated during a minimum of 10% of dwells and 10 minutes of sweeps. Note – The 15 and 46 Hertz modes arise when the vertical tail is in buffet, generally above 20° degrees AOA.	Rudder Sinusoidal Vibration Cyclic sweeps from 5 to 85 to 5 Hz with 10 minutes per sweep and with dwells at two most severe resonances. Total test time is 20 minutes per axis, including 5 minute dwells at two most severe resonance points. Check accuracy at end of test. Conduct test as defined for surface neutral. For the fully extended and fully retracted positions, conduct a single sweep with 2 minute dwells at the two most severe resonance points. For each ram position, conduct narrow band sweeps in the lateral axis over 13 to 17 Hertz and over 40 to 52 Hertz per Curve II of Figure 7 (Reference: Acceptance and Flight Worthiness Test Plan for FACT Program Linear Sensors, Revision A). Sweep duration is: 5 minutes at surface neutral, 2 minutes at fully retracted and 2 minutes at fully extended ram positions. This test can be waived at a particular ram position and frequency band if a dwell is conducted at a resonance within that band and at that ram position in the previous step.
	Random Vibration – 120 minutes vibration per prescribed spectra per axis. Performance verified for minimum of 30 minutes of test. Breakdown for stabilator: 104 minutes at surface neutral and 4 minutes at each of 3.2 inch extended, mid-stroke, 1.8 inch retracted and 3.2 inch retracted. F-18 E/F uses 60 minutes at surface neutral and 20 minutes at fully extended, midstroke and fully retracted ram positions each. Note: Testing at "performance" levels for indicated times qualifies for a 50-hour block of flight. Qualification for a 6,000 hour life can be achieved by testing at the "endurance" levels.	Random Vibration – 30 minutes vibration per axis using modified spectra with overall g rms. = 26.1 in lateral axis and 24.8 in vertical and longitudinal axes (lower than F-18 specified). Test for 20 minutes at surface neutral and 5 minutes at fully extended and fully retracted ram positions each.
	Final Check – Repeat Threshold, EHV Null Bias, Frequency Response, External Leakage and Internal Leakage portions of the Acceptance Tests	Final Check – Conduct actuator ATP before and after vibration tests and compare results.

The sensor passed the life cycle wear, pressure impulse cycling, temperature, and altitude tests. The original Life Cycle tests revealed wearing of the sensor mask causing fine debris in the sensor. An improved process of creating the sensor mask eliminated the wear, and the second life cycle test showed no wear or other problems.

The sensor was installed in the actuator for the vibration tests consisting of sinusoidal resonance survey, sinusoidal dwell and cycling, and random vibration. No rudder sensor or actuator failures occurred. The sinusoidal and random vibration profiles are shown in figures 29 through 32. The vibration profiles were taken from the F/A 18 rudder actuator profiles with the random vibration profiles for the lateral, longitudinal, and vertical axes modified to decrease the input acceleration power spectral density (g_{rms}^2/Hz). The overall g_{rms} values were reduced from the original F/-18 specifications: 35.0 g_{rms} to 25.1 g_{rms} for the lateral axis and from 27.9 g_{rms} to 24.8 g_{rms} for the longitudinal and vertical axes. The reduced values better matched the sensor requirements specified at the start of the program. Since the random vibration test levels were reduced, the FACT rudder actuator load limit is 6.0 g instead of the usual 7.5 g. The envelope for NASA-Dryden's Systems Research Aircraft (SRA) was not reduced since the SRA already has a 6.0 g placard.

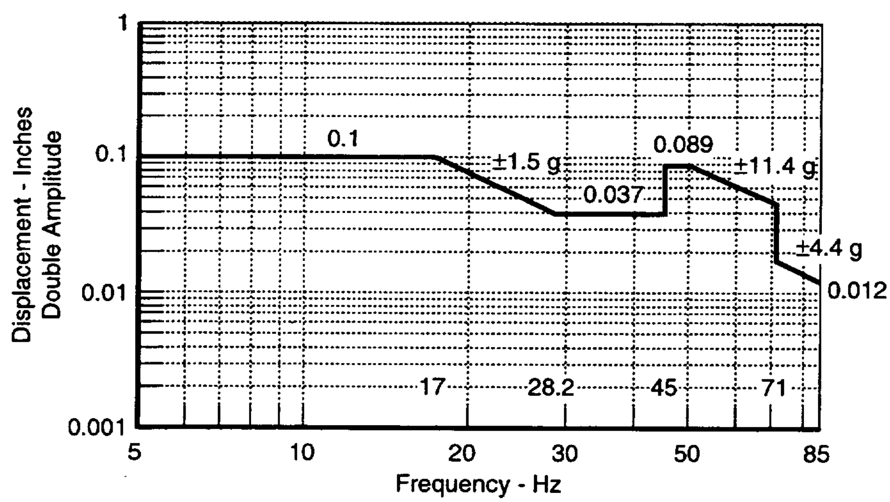
The sinusoidal resonance survey was a ten minute sinusoidal sweep from 5 hertz to 2000 hertz of the lesser of 0.024 inch double amplitude displacement or ±2g. Table 4 summarizes the resonances for each axis and actuator stroke position.



Note: Performance and Endurance Levels Coincide Below 85 Hz

GP71485065.cvs

Figure 29. Rudder Sinusoidal Vibration for Lateral Axis



Note: Performance and Endurance Levels Coincide Below 85 Hz

GP71485066.cvs

Figure 30. Rudder Sinusoidal Vibration for Vertical and Longitudinal Axes

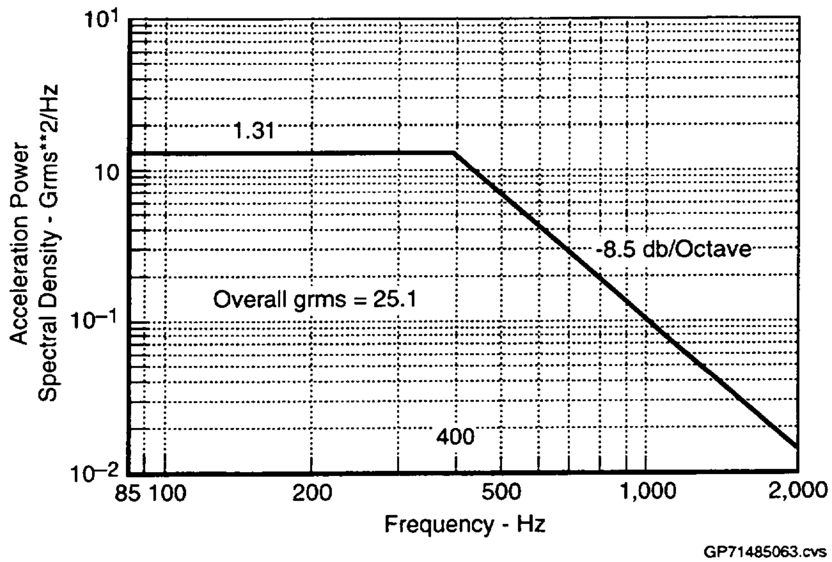


Figure 31. Rudder Random Vibration for Lateral Axis

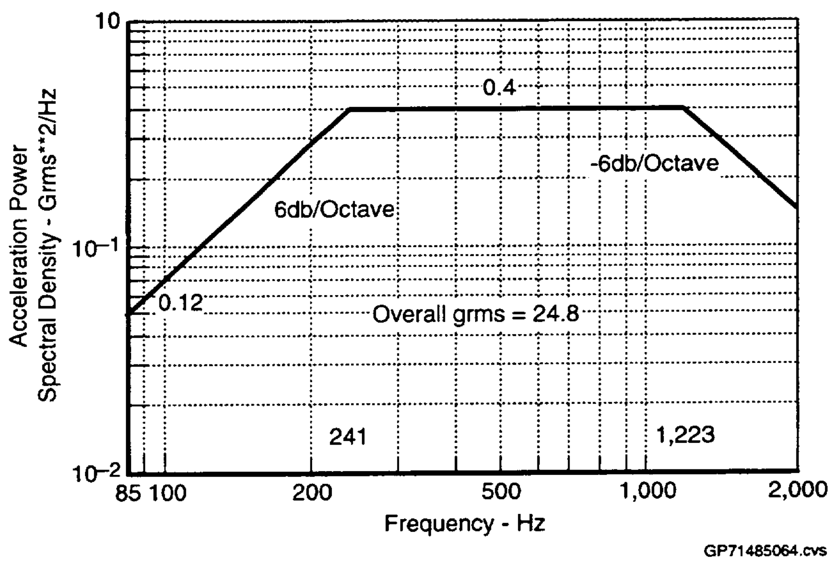


Figure 32. Rudder Random Vibration for Vertical and Longitudinal Axes

TABLE 4. ACTUATOR RESONANCE SURVEY SUMMARY

Axis	Actuator Position	Resonances Frequency (Hz)	Level (g)
Longitudinal	Midstroke	401.9	4.1
	Fully Retracted	329.3	4.6
	Fully Extended	none	none
Lateral	Midstroke	98.0	20.6
	Fully Retracted	77.1	6.2
	Fully Extended	94.2	5.5
Vertical	Midstroke	77.1	6.8
	Fully Retracted	106.1	16.9
	Fully Extended	99.3	13.8

Sensors were not installed in stabilator actuators due to problems assembling the sensors. Humidity, contamination, and optic fibers breaking caused several failures in making acceptable optic sensors for the stabilator actuator. After the sensor assembly process was correct, difficulty installing the sensor in the stabilator cylinder caused more schedule delays. A stabilator sensor was completed, but it was too late to proceed with the stabilator portion of the FACT program.

7.3 EOA Module Acceptance Tests

AlliedSignal initially performed the EOA module acceptance tests according to the Standard Hardware Acquisition and Reliability Program (SHARP) requirements, but the EOA failed to pass the airworthiness vibration tests. The vibration levels were then reduced to match those used for the ICU Airworthiness vibration tests, and the EOA passed. While the EOA did not meet the SHARP common module requirements, it did meet the FACT program requirements.

7.4 Power Supply Module Acceptance Tests

The Naval Air Warfare Center (NAWC) Indianapolis SHARP performed the power supply acceptance tests per MIL P 29590 paragraph 4.7.5.2 with the following limits: the vibration profile duration was 10 minutes in the axis perpendicular to the module; the thermal test consisted of 12 cycles of 5 hours each with minimum temperature of -55°C and maximum temperature of 70°C.

7.5 Environmental Stress Screening Tests

Vibration and temperature environmental stress screening tests were performed on all modules to eliminate weak components before being integrated into an ICU or FRU. Module level tests were requested by NASA-Dryden to screen the commercial components used in the FACT system, and to verify the ruggedness of the modules. The tests were based on recommendations in NAVMAT P-9492 (Navy Manufacturing Screening Program). That document pertains to box level tests for production quantities or long life equipment, but the information was tailored to fit the needs of the FACT program. Figure 33 shows the vibration profile and Figure 34 shows the temperature profile for environmental stress screening of the Interface and Feedforward modules. The modules were not operating during these tests.

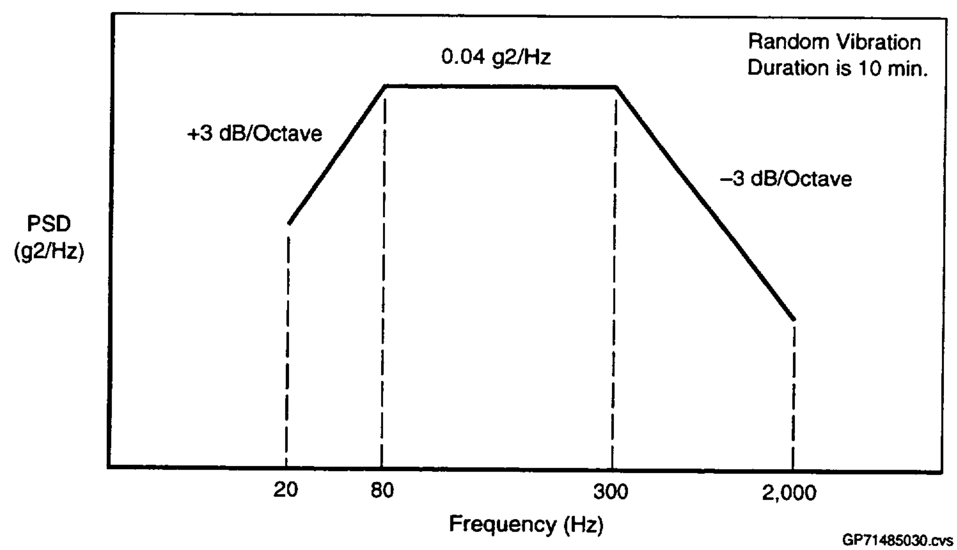


Figure 33. Vibration Stress Screening Profile

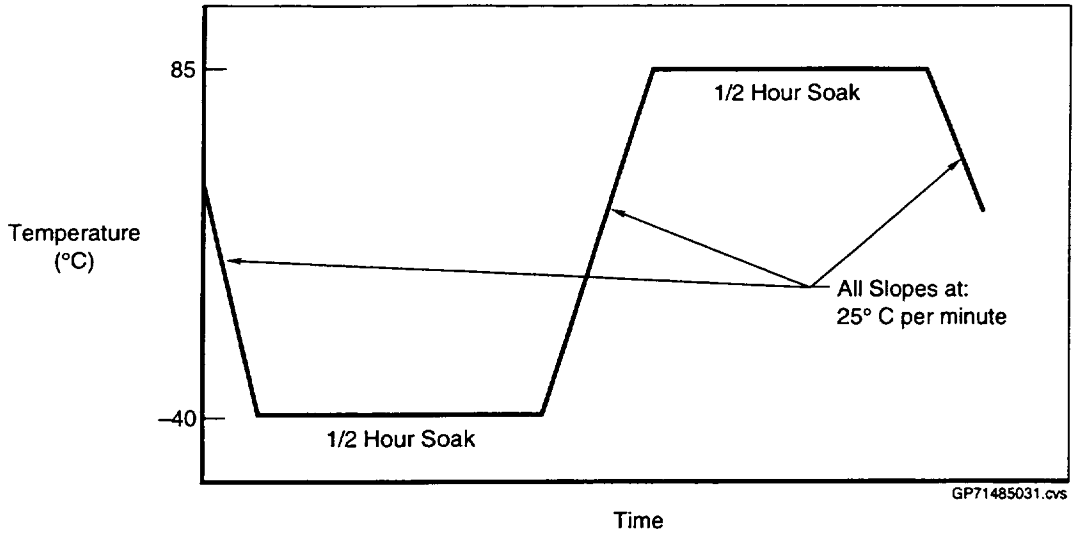


Figure 34. Temperature Stress Screening Profile

The environmental stress screening tests are less stressful than the environmental airworthiness tests but provided confidence in the ruggedness of the modules. One problem was found during the interface and feedforward module environmental stress screening tests performed at Boeing St. Louis. On one interface module, one lead on a large capacitor broke in the vibration test. The capacitor was replaced, extra epoxy was applied, and the module passed the retest. Additional epoxy was added to secure the large capacitors on all interface modules.

7.6 Component Tests

The component tests verified the operation of all of the components and functions of the ICUs and FRUs. Each function was tested by varying its inputs and checking its outputs against expected results. The component tests were partitioned into power switching tests using the ICU, command feedforward tests using the ICU and FRU, and position feedback tests using the ICU and optic sensor or actuator.

7.6.1 Power Switching Function

The power switching function of the interface module was tested by reducing the voltage of the ICU main 28 volt power input and verifying the switch to battery 28 volt power. The reduced main power input was then increased to 28 volts and the switch from battery to main 28 volt power was verified. The ICU power supply is a Navy SHARP standard module and was built and tested to comply with MIL-STD-704A.

Table 5 contains the power switching test results. No power switching failures occurred.

TABLE 5. POWER SWITCHING COMPONENT TEST SUMMARY

POWER SWITCHING COMPONENT TEST SUMMARY	MAIN TO BATTERY POWER SWITCHING SYSTEM (ICU with Main and Battery 28V Inputs)					
TEST	1	2	3	4	5	6
Voltage at Switch from Reducing Main Power to Battery Power Expect: 16.8Vdc +/- 1.0Vdc	17.4 Vdc	17.3 Vdc	16.4 Vdc	17.5 Vdc	17.3 Vdc	17.3 Vdc
Time on Battery Power Before Switch Back to Main Power Expect: 8.6sec +/- 0.5sec	9.0 sec	9.0 sec	8.9 sec	8.9 sec	9.0 sec	9.1 sec
Voltage at Switch from Battery Power to Increasing Main Power within 8.6 seconds Expect: 18.6V +/- 0.5V	18.6 Vdc	18.6 Vdc	18.4 Vdc	18.6 Vdc	18.5 Vdc	18.5 Vdc

The power switching function performed as expected. Provided the FACT system is supplied with main 28 volt and battery 28 volt power, the power switching function provides the FACT system power supply with a backup power source for up to nine seconds. See Development section 6.2.5.1, IM Power Switching, for an explanation of the power switching function.

7.6.2 Feedforward Command Function

The feedforward command path functions were tested using the feedforward modules of the ICU and FRU to keep the optic transmitters and receivers communicating with each other so no special optic function generators were needed. Command versus actual current linearity was tested by injecting a known command into the ICU and comparing it to the current through the actuator and FRU. The feedforward instrumentation interface for data acquisition was also checked during the linearity test. Failure monitors were tested by creating failures, electrical opens and shorts at the actuator control valve and open optic links to and from the actuator, and verifying the failures were detected at the proper thresholds and times; the return to normal operation was also tested. Feedforward performance was tested with a frequency response, optic power margin test, and a noise amplitude test.

Table 6 contains the feedforward command test results. Test anomalies are discussed in the following paragraphs. Anomalies are judged by the test plan expected results and can be due to incorrect expected results or equipment fails. See Development sections 6.2.4, 6.2.4.1, and 6.2.4.2 for the Feedforward Module for an explanation of the feedforward command function.

Except for the following, the feedforward command function performed as expected.

1. The test point voltages recorded in the command versus current least squares line fit section were essentially within the expected values for all feedforward systems. Only two data points were out of the expected range, and the points were only out of range by 0.1% and 0.3%. The small errors were determined to be acceptable.

2. The command versus current monitor for detecting an inconsistency in the command between the ICU and FRU failed by the error detection threshold being greater than expected in four of the six systems tested. For the following reasons, no action was taken to change the Feedforward Module to achieve the 3.21 milliampere trip level as originally specified, and the expected results were changed to a range of 3.2 through 3.85 milliamperes. The feedforward monitor still tripped at a value less than the full FCC servo-amplifier output, four milliamperes. Full FCC servo-amplifier output is achieved with a rudder command versus position error of less than two degrees. So, the FACT command versus current monitor will detect an error unless the surface is within two degrees of the commanded position. In addition, there is an FCC monitor of the actual EHV spool position independent of the FACT system. This monitor trips when the EHV spool position is not within 0.02 inches of its expected value; full spool travel is $\pm$ 0.03 inches. The spool is a hydraulic porting device and an intermediate stage between the EHV, which receives the FCC command, and the main ram, which moves the control surface.

3. The EHV current noise was greater than expected for all feedforward systems, but the largest noise level under 1 kHz translates into less than 0.8 degrees rudder surface. The test is not good for indicating the performance of the FACT system. The test limit is arbitrary and has not been tied to FACT system performance. Also, the quality of the measurement depends on the quality of the ground reference which will be different at the aircraft, Iron Bird, and even different laboratories. The test was left in the procedures to get an example of noise levels in a less than ideal laboratory environment.

4. The frequency response gain drops 3 dB below the zero reference value at 80 Hertz instead of beyond 1000 Hertz and reaches a maximum of -6 dB near 320 Hz. The expected gain results are based on a purely resistive load rather than an EHV coil and are incorrect when an actuator is used for the load. The EHV coil load specification is 6 henries, 1000 ohms. When the impedance of the EHV coil is taken into account, the test results are as expected. The expected corner frequency is 26.5 Hertz, $1000\Omega/(6H \times 2\pi)$. Frequency = $\omega/2\pi = R/(L2\pi)$. The actual corner frequency is 20 Hertz and near the low end of the expected range when factoring in the resistance and inductance tolerances.

TABLE 6. COMMAND FEEDFORWARD COMPONENT TEST SUMMARY

COMMAND FEEDFORWARD COMPONENT TEST SUMMARY	RUDDER FEEDFORWARD SYSTEM (ICU, FRU, and Actuator)					
TEST	1	2	3	4	5	6
POWER UP and RESET						
ICU and FRU Interrupt Relays are Passing Signal After Power Up	✓	✓	✓	✓	✓	✓
COMMAND vs CURRENT LEAST SQUARES LINE FIT						
Slope Expect: 1.00 ± 0.05	1.008	1.001	0.996	1.002	0.995	0.995
Null Offset Expect: ≤ 0.16 mA	0.004mA	0.014mA	-0.019mA	-0.008mA	0.012mA	-0.003mA
Maximum Deviation of Results from Best Fit Line Expect: ≤ 0.16mA	-0.0083mA	-0.0044mA	0.0042mA	-0.0044mA	0.0077mA	-0.0044mA
Test Point Voltages Within 2% of Calculated Value	-4mA value 0.1% low	✓	2mA value 0.3% low	✓	✓	✓
COMMAND vs CURRENT MONITOR OPERATION for FAILED EHV CURRENT						
Centertap Voltage Shows Fail for 5 seconds Expect: < 7.4 Vrms	✓	✓	✓	✓	✓	✓
Threshold Current When Monitor Detects Fail Then Opens Relays Expect: 3.21 mA $\pm$ 2%	3.79 mA	3.68 mA	3.34 mA	3.36 mA	3.26 mA	3.22 mA
ICU and FRU Relays Remain Open After 5 seconds After Fail	✓	✓	✓	✓	✓	✓
RESET AFTER MONITOR FAIL						
Centertap Voltage Shows No Fail After 5 seconds After Fail Expect: 7.7 Vrms $\pm$ 0.3 Vrms	✓	✓	✓	✓	✓	✓
ICU and FRU Relays are Closed	✓	✓	✓	✓	✓	✓
RESPONSE OF MONITOR WITH A LARGE COMMAND CURRENT AND AN OPEN EHV						
Centertap Value Is < 7.4 Vrms Within 25ms of Opening EHV	✓	✓	✓	✓	✓	✓
ICU and FRU Relays Remain Open After 5 seconds After Fail	✓	✓	✓	✓	✓	✓
RESPONSE TO OPTIC FAILURE						
With Optic Fail in Fiber Carrying FCC Command, Centertap Voltage Shows Fail for 5 seconds Expect: < 7.4 Vrms	✓	✓	✓	✓	✓	✓
With Optic Fail in Fiber Carrying EHV Current, Centertap Voltage Shows Fail for 5 seconds Expect: < 7.4 Vrms	✓	✓	✓	✓	✓	✓
ICU and FRU Relays Remain Open After 5 seconds After Fail	✓	✓	✓	✓	✓	✓
FEEDFORWARD PERFORMANCE						
Optic Power Margin of Position Expect: > 3 dB	11.2 dB calculated	7.5 dB calculated	21.5 dB	19.09 dB	18.6 dB	17.2 dB
Voltage of EHV Current Noise (Using 1kHz cutoff Low Pass Filter) Expect: ≤ 16 mV	0.5V noise (no filter used)	0.5V noise (no filter used)	52mV maximum	140mV maximum	56mV maximum	32mV maximum
Frequency Response of FCC Command vs EHV Current Expect: -3dB < gain < 0.5dB $-90^\circ < \text{phase} \leq 0^\circ$	gain -6.0dB at 331Hz phase OK	gain -5.9dB at 331Hz phase OK	gain -5.4dB at 316Hz phase OK	gain -5.4dB at 316Hz phase OK	gain -5.4dB at 320Hz phase OK	gain -5.4dB at 302Hz phase OK
Voltage of FCC Command vs EHV Current is < 2.4 Vdc Over the Full Sensor Range	✓	✓	✓	✓	✓	✓
DATA ACQUISITION INTERFACE						
Data Acquisition Interface Works	✓	✓	✓	✓	✓	✓

The optic power margin was measured by connecting the system shown in Figure 35 with the variable attenuator set at minimum attenuation (as read with an optic power meter attached to the connectors mating to the ICU and FRU front panels). The attenuation was increased until the system failed. The attenuator was removed from the system and measured. The difference between the first and last attenuator measurement is the optic power margin.

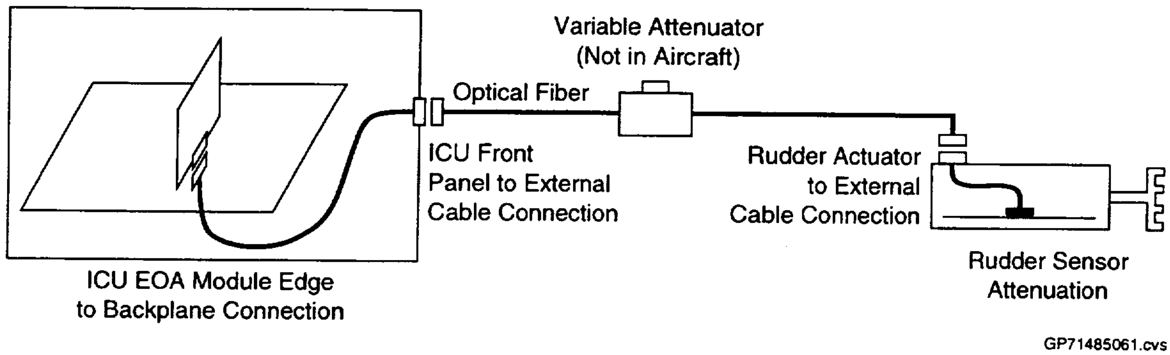


Figure 35. Feedforward System Power Margin Test Setup

Throughout the feedforward component tests, the feedforward systems performed solidly. Performance was as expected and consistent from day to day and from system to system. The feedforward function started operating without fails after power up and reset. The command input versus actual current output was very linear. Failure detection was unfailing; the command versus actual current monitor detected failures, and the feedforward system detected optic signal failures. The system was consistently reset after failures were removed. The optic power margin was very high, about 20 dB. The component test results show the feedforward system is a good system.

7.6.3 Feedback Position Function

The feedback position path functions were tested using the EOA module of the ICU and the optic sensor to keep the optic transmitters, sensor, and receiver communicating with each other so no special optic function generators were needed. The feedback outputs during power up and reset were tested for the timing of proper operation and fail indications. The position was tested over its full range for correct optical decoding and signal phase. Accuracy checks were made at full extend and retract positions. An optic power margin test was performed, and the instrumentation interface to the data acquisition system was checked to ensure all data is transmitted and agrees with the current position and operational status reported to the FCC. Failure monitors were tested by creating failures, open paths between the optic transmitter and sensor and between the optic sensor and receiver, and verifying the failures were detected at the proper times; the return to normal operation was also tested.

Table 7 contains the feedback position test results. Test anomalies are discussed in the following paragraphs. Anomalies are judged by the test plan expected results and can be due to incorrect expected results or equipment fails. See Development section 6.2.3 for the Electro-Optic Architecture Module (EOA) for an explanation of the feedback position function.

Except for the following, the feedback position function performed as expected. Most of the anomalies were in the position decoding tests.

1. The position voltages at sensor full extend for feedback system one and two were higher than allowed. Since this same test was successfully executed as part of the vendor acceptance test on the modified rudder and again in the Boeing system test, the result is due to the sensor which was an early prototype sensor. The system test results at full extend in both channels was 5.306 Vrms and within 5.376 $\pm$ 2%. The optic position sensor used with system one and two component tests had failed vendor sensor tests due to differences between the channels so it could not be used for flight. The sensor could still be used to create a valid sensor signal to one feedback channel and was used because it was the only sensor available.

TABLE 7. POSITION FEEDBACK COMPONENT TEST SUMMARY

POSITION FEEDBACK COMPONENT TEST SUMMARY	RUDDER FEEDBACK SYSTEM (ICU and Optic Sensor in Actuator or Stand Alone)					
TEST	1	2	3	4	5	6
POWER UP and RESET						
Centertap Voltage within 5 seconds of Power Up or Reset Expect: < 7.4 Vrms	0.300 Vrms	0.380 Vrms	0.240 Vrms	0.220 Vrms	0.240 Vrms	0.220 Vrms
Rudder Position Voltage within 5 seconds of Power Up or Reset Expect: < 7.4 Vrms	0.002 Vrms	0.003 Vrms	0.260 Vrms	0.220 Vrms	0.260 Vrms	0.220 Vrms
Centertap Voltage after 5 seconds of Power Up or Reset Expect: 7.7 Vrms ± 0.3 Vrms	7.64 Vrms	7.94 Vrms	7.96 Vrms	7.80 Vrms	7.86 Vrms	7.78 Vrms
POSITION DECODING TESTS						
Position Voltage at Full Extend Expect: 5.376Vrms ±2% out phase	5.721Vrms ϕ = 181.1 °	5.670Vrms ϕ = 181.1 °	5.410Vrms ϕ = 179.9 °	5.396Vrms ϕ = 179.8 °	5.369Vrms ϕ = 179.7 °	5.365Vrms ϕ = 180.0 °
Position Voltage at Full Retract Expect: 5.376Vrms ±2% in phase	5.387Vrms ϕ = 359.5 °	5.337Vrms ϕ = 359.5 °	5.403Vrms ϕ = 359.9 °	5.423Vrms ϕ = 359.8°	5.385Vrms ϕ = 359.7 °	5.417Vrms ϕ = 359.7 °
Phase Angle Over Entire Position Range Expected: < 5 °	>5 ° from 45mVrms in phase to 45mVrms out phase	>5 ° from 45mVrms in phase to 45mVrms out phase	>5 ° from 13mVrms in phase to 8mVrms out phase	>5 ° from 48mVrms in phase to 49mVrms out phase	>5 ° from 49mVrms in phase to 58mVrms out phase	>5 ° from 55mVrms in phase to 51mVrms out phase
Smallest Centertap Voltage Over Entire Position Range Expect: 7.7 Vrms ± 0.3 Vrms	7.84 Vrms	7.92 Vrms	7.86 Vrms	7.78 Vrms	7.76 Vrms	7.76 Vrms
Largest Centertap Voltage Over Entire Position Range Expect: 7.7 Vrms ± 0.3 Vrms	7.96 Vrms	8.04 Vrms	7.92 Vrms	7.88 Vrms	7.96 Vrms	7.82 Vrms
RESPONSE TO OPTIC FAILURE						
Centertap Voltage with Optic Fail in Fiber Carrying Source Expect: < 7.4 Vrms	0.100Vrms	0.460Vrms	0.300Vrms	0.280Vrms	0.240Vrms	0.280Vrms
Centertap Voltage with Optic Fail in Fiber Carrying Signal Expect: < 7.4 Vrms	0.300Vrms	0.480Vrms	0.300Vrms	0.280Vrms	0.260Vrms	0.280Vrms
Rudder Position Frozen with Optic Fail in Source or Signal Fiber	✓	✓	✓	✓	✓	✓
RECOVERY FROM OPTIC FAIL						
Centertap Voltage after 5 seconds of Removing Fail in Source Fiber Expect: 7.7 Vrms ± 0.3 Vrms	7.64 Vrms	7.94 Vrms	7.96 Vrms	7.84 Vrms	7.80 Vrms	7.80 Vrms
Centertap Voltage after 5 seconds of Removing Fail in Signal Fiber Expect: 7.7 Vrms ± 0.3 Vrms	7.64 Vrms	7.94 Vrms	7.96 Vrms	7.82 Vrms	7.82 Vrms	7.82 Vrms
Rudder Position is Current Position	✓	✓	✓	✓	✓	✓
OPTIC POWER MARGIN						
Optic Power Margin of Position Expect: > 3 dB	0.75 dB calculated	3.01 dB calculated	9.30 dB	5.99 dB	10.74 dB	5.63 dB
DATA ACQUISITION INTERFACE						
Data Acquisition Interface Works	✓	✓	✓	✓	✓	✓

2. During sensor movement in all feedback systems, the phase angle of the position feedback relative to excitation voltage exceeds five degrees at voltages less than ±60 mVrms, near zero. Since there was no actuator chatter during the rudder ATP tests at Dowty, and no oscilloscope waveform movements of position compared to excitation during the feedback component tests, the failure about zero Vrms is due to the ability of the phase angle volt meter to resolve the phase angle with at a low amplitude signal. For another check on this fail, the phase relationship near the null position (i.e., small amplitude) is indirectly verified by three successful small amplitude frequency responses in the weight off wheels section of the system test.
3. The largest rudder centertap voltage in feedback system two was 8.04 Vrms, 0.04 Vrms over limit. This fail is not a concern for two reasons. The FCC specification limit is 7.7±0.3 Vrms, but the limit of the actual FCC circuit measuring the rudder centertap voltage is 6.88±5% to 8.81±5% Vrms so 8.04 Vrms is not a fail. Also, test equipment is probably the cause of the fail. The rudder centertap voltage output is directly proportional to the excitation voltage that was created with a function generator whose output varied by about 0.1 Vrms. The varying excitation voltage is the reason the rudder centertap voltages vary between feedback systems.
4. The optic power margin of feedback system one was calculated to be 0.75 dB. The calculated optic power margin was found to be inaccurate. After component tests for feedback systems one and two, an accurate method was found to measure the optic power margin. Calculated optic power margins for feedback systems three through six were 0.00 to 1.51 dB, but measured optic power margins were 5.63 to 10.74 dB. Thus, optic power margins for feedback systems one and two are much larger than the calculated value and are above the 3 dB limit.

The optic power margin was measured by connecting the system shown in Figure 36 with the variable attenuator set at minimum attenuation (as read with an optic power meter attached to the connectors mating to the ICU front panel and rudder actuator). The attenuation was increased until the system failed. The attenuator was removed from the system and measured. The difference between the first and last attenuator measurement is the optic power margin.

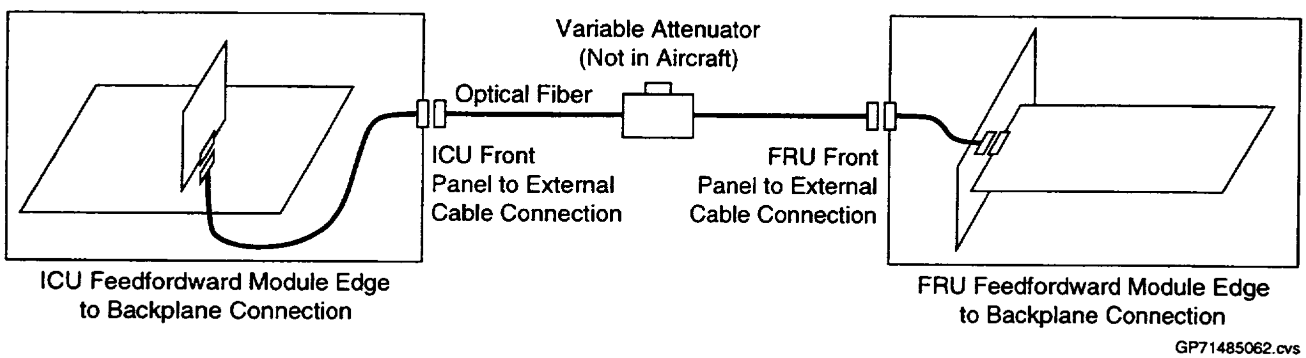


Figure 36. Feedback System Power Margin Test Setup

Throughout the feedback component tests, the feedback systems performed solidly. Performance was as expected and consistent from day to day and from system to system. The feedback function started operating without fails after power up and reset. The position decoding was accurate. The feedback system detected optic signal failures and was consistently reset after failures were removed. The optic power margin of 5.6 dB to 10.7 dB was adequate. The component test results show the feedback system is a good system.

7.6.4 Component Test Summary

Initial component testing revealed some failed electrical components, an inadequate optic transmitter problem in the feedforward modules, and a logic design problem in the interface module. The feedfoward problem was corrected by replacing the optic transmitter with a faster switching transmitter. The interface module logic was corrected with electronic trace cuts and jumpers, and the failed components were replaced. All of the affected modules repeated the environmental stress screening after these changes.

After the resolution of the initial problems, the feedforward and feedback components performed extremely well in component tests. The components met almost all of the expected results with some minor failures determined to be acceptable. The components operated many hours without failures, showed consistent results when tests were repeated, and showed no unusual or undefined problems. One set of components was ready for system tests and then delivery to NASA-Dryden, and the other set of components was ready for airworthiness tests.

7.7 System Tests

The system tests verified proper operation of the FACT system (FCC, ICU, FRU, and actuator) in a production system environment (FCC test station supplying FCCs with aircraft information and hydraulic bench powering the actuator). The FACT system test setup is shown in Figure 37. The system tests were partitioned into functional tests with the aircraft on the ground, functional tests with the aircraft in the air, system performance, and failure modes and effects tests.

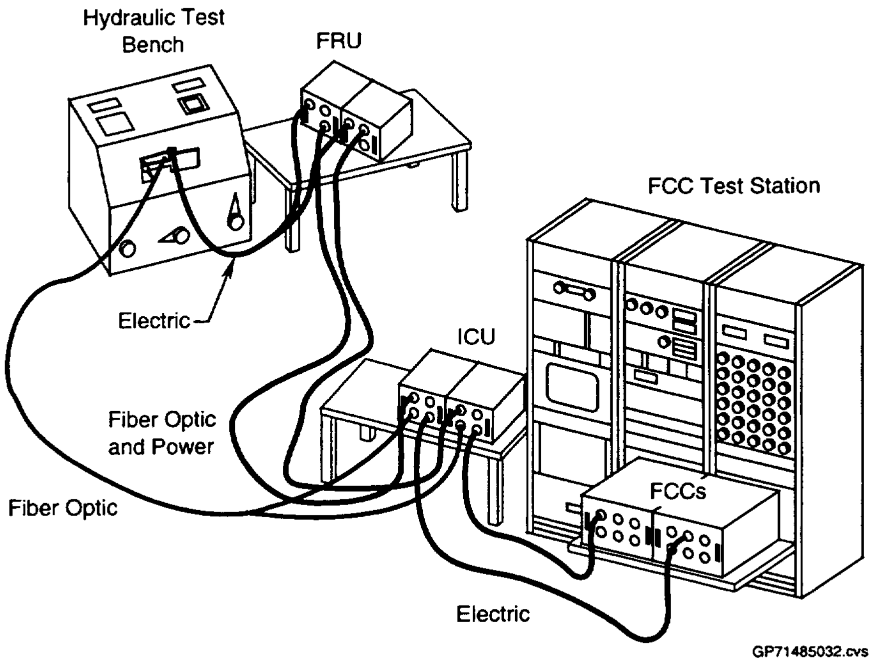


Figure 37. Laboratory Test Setup

7.7.1 System Functional and Performance Tests

The functional tests with the aircraft on the ground checked for proper operation when power was applied to the system, during the FCC built-in-test (BIT), when one and two FCC channels controlled the actuator, during system reversion to backup, and during system recovery from backup to normal operation.

The following summarizes the aircraft weight on wheels tests. The integrated system powered up correctly no matter if the FACT system or FCS system was turned on first. The integrated system passed the initiated built-in-test run by the pilots to test the system. The rudder operation was normal and showed no chatter for one and two FCC channel operation. Simulated cockpit switches successfully shut off the FACT rudder system leaving the rudder in trail damp mode, FCS backup mode, and successfully reset the system leaving the rudder in CAS mode, FCS normal mode.

The functional tests with the aircraft in the air checked for proper operation when power was applied to the system, when one and two FCC channels controlled the actuator, during system reversion to backup, during system recovery from backup to normal operation, during actuator reversion and recovery transients which were checked for severity, and when main power was interrupted.

The following summarizes the aircraft weight off wheels tests. The FACT system powered up correctly with the FCS system already on. The rudder operation was normal and showed no chatter for one and two FCC channel operation.

Simulated cockpit switches successfully shut off the FACT rudder system leaving the rudder in trail damp mode, FCS backup mode, and successfully reset the system multiple times leaving the rudder in CAS mode, FCS normal mode. The FACT system successfully recovered from one and two channel failures in the feedforward function and feedback function. The battery backup switching function successfully provided backup power when primary power was lost, and the rudder position showed no anomalies during the switch. The rudder system performance was very good. The EHV current and actuator position tracking between the two FCC channels controlling the actuator were much less than allowed. The optic sensor feedback and electric conversion to an LVDT like signal was accurate; the electric signal correctly corresponded to the actuator position, the position data was close to the command versus position best fit line, and there was little hysteresis. The actuator started and stopped moving with a small amount of command. The rudder system command versus position large and small input amplitude frequency responses were within the acceptance test plan limits for the production fly-by-wire system.

The system performance tests checked the FACT system servo-loop performance against the production system performance. Tests compared the tracking of actuator control valve currents between the two computer channels controlling the rudder actuator, determined the threshold of command to create and stop actuator motion, measured hysteresis in actuator movement, plotted large and small source amplitude frequency responses for one and two controlling channels, and compared accuracy, tracking, null offset, and phase shift of the actuator position to the production sensor, linear variable differential transformer (LVDT), specifications.

The performance of the FACT system integrated into the flight control system was superb and like the production fly-by-wire system.

Table 8 contains the system functional and performance test results. The test anomalies are discussed in the following paragraphs. See System Description section 6.1.1 for the System Overview for an explanation of the FACT system.

1. Rudder system one failed the maximum deviation from the best fit line in the sensor feedback section of the rudder system performance tests. One data point in the command versus position data deviated from the best fit line by 0.0036 Vrms. The deviation is acceptable for two reasons. The deviation is relatively small at 14%, and the next worse data point deviation is 18 mVrms, 0.007 Vrms under the limit. To make sure no problem exists, NASA-Dryden will retest linearity to verify the data gathered in this test; similar or better linearity will be acceptable.

2. Rudder system two needed an extra ICU reset to clear a fault after power up of the FACT units then FCC in the FACT system power up section of the aircraft weight on wheels tests. While the system is expected to power up in the operational state, requiring a reset after power up to achieve the operational state is just an extra step in the power up process. The flight control system (FCS) handles an unpowered system as a failed channel and does not use that channel until an FCS reset is manually pressed to attempt to engage the channel. Using the ICU reset to obtain an operational state for an ICU channel is a natural step before using an FCS reset to engage the channel. The FCS reset prevents the FACT system from powering up in a failed state and affecting the flight control system.

3. Rudder system two needed an extra flight control system reset to clear a fault while recovering from a two channel failure in the recovery from failures section of the aircraft weight on wheels tests. This discrepancy is an unexpected event during system testing and not a failure. Multiple FCS resets are acceptable to engage a channel.

The FACT system performed extremely well when integrated into the flight control system (FCS). The FCS with the FACT system, the integrated system, behaved like the production fly-by-wire system with the FCS on the ground, aircraft weight on wheels, or in the air, aircraft weight off wheels.

TABLE 8. RUDDER INTEGRATED SYSTEM TEST SUMMARY (PART 1)

RUDDER INTEGRATED SYSTEM TEST SUMMARY	RUDDER SYSTEM (ICU, FRU, Actuator, and FCC for Channels 1 & 4)			
	SYSTEM 1		SYSTEM 2	
TEST	CHANNEL 1	CHANNEL 4	CHANNEL 1	CHANNEL 4
AIRCRAFT WEIGHT ON WHEELS TESTS				
POWER UP FACT SYSTEM FOLLOWED BY FLIGHT CONTROL COMPUTER (FCC)				
AFTER FACT POWER UP				
Feedforward Channels in Fully Operational State	✓	✓	✓	✓
Feedback Channels in Fully Operational State	✓	✓	✓	✓
AFTER FCC POWER UP				
Left and Right Rudders are in CAS Mode	✓		✓	
POWER UP FCC FOLLOWED BY FACT SYSTEM				
AFTER FCC POWER UP				
Left Rudder in Trail Damp Mode	✓		✓	
Right Rudder in CAS Mode	✓		✓	
AFTER FACT POWER UP				
Feedforward and Feedback Channels are Operational	✓	✓	✓	✓ Needed ICU Reset
Centertap Voltage Expect: 7.64 ± 0.3 Vdc	7.56 Vdc	7.59 Vdc	7.53 Vdc	7.54 Vdc
Left Rudder Remains in Trail Damp Mode	✓		✓	
After FCS Reset, Left Rudder is in CAS Mode	✓	✓	✓	✓
FLIGHT CONTROL SYSTEM (FCS) INITIATED BUILT-IN-TEST (IBIT)				
FACT System Has No Effect on IBIT	✓		✓	
FCC Passes IBIT	✓		✓	
RUDDER FUNCTIONAL TESTS				
CHANNEL 1 & 4 OPERATIONAL				
Centertap Voltage Expect: 7.64 ± 0.3 Vdc	7.54 Vdc	7.56 Vdc	7.54 Vdc	7.53 Vdc
Voltage Difference Between CH1 & CH4 Expect: ≤0.06 Vdc	0.01 Vdc		0.02 Vdc	
Rudder Operates Without Chatter or Instability	✓		✓	
CH 1 OPERATIONAL, CH 4 DISABLED				
Rudder Operates Without Chatter or Instability	✓		✓	
CH 4 OPERATIONAL, CH 1 DISABLED				
Rudder Operates Without Chatter or Instability	✓		✓	
MOMENTARILY SWITCHING LEFT RUDDER CAS OFF (SIMULATE COCKPIT SWITCH)				
Left Rudder Reverts to Trail Damp Mode	✓		✓	
Right Rudder Remains in CAS Mode	✓		✓	
After FCS Reset, Left Rudder is in CAS Mode	✓	✓	✓	✓
INTERFACE CONVERTER UNIT (ICU) RESET				
Rudder Position is 0 Vdc for 5 sec.	✓	✓	✓	✓
Centertap Voltage is ≤6.4 Vdc for at least 5 sec.	✓	✓	✓	✓
Left Rudder Reverts to Trail Damp Mode	✓		✓	
After FCS Reset, Left Rudder is in CAS Mode	✓	✓	✓	✓
BATTERY BACKUP SWITCHING				
No Rudder Position Anomaly with Removing Main 28V Power in CH 1 or CH 4 for < 9 seconds.	✓	✓	✓	✓
No Rudder Position Anomaly with Removing Main 28V Power in CH 1 & CH 4 Simultaneously for <9 seconds.	✓		✓	

TABLE 8. RUDDER INTEGRATED SYSTEM TEST SUMMARY (PART 2)

RUDDER INTEGRATED SYSTEM TEST SUMMARY (continued)	RUDDER SYSTEM (ICU, FRU, Actuator, and FCC for Channels 1 & 4)			
	SYSTEM 1		SYSTEM 2	
TEST	CHANNEL 1	CHANNEL 4	CHANNEL 1	CHANNEL 4
AIRCRAFT WEIGHT OFF WHEELS TESTS				
POWER UP FACT SYSTEM WITH FCC ALREADY POWERED				
Feedforward Channels in Fully Operational State	✓	✓	✓	✓
Feedback Channels in Fully Operational State	✓	✓	✓	✓
Centertap Voltage is 7.7 ± 0.3 Vrms	✓	✓	✓	✓
Left Rudder Remains in Trail Damp Mode	✓		✓	
RUDDER FUNCTIONAL TESTS				
CHANNEL 1 & 4 OPERATIONAL				
Centertap Voltage Expect: 7.64 ± 0.3 Vdc	7.55 Vdc	7.57 Vdc	7.53 Vdc	7.53 Vdc
Voltage Difference Between CH1 & CH4 Expect: 0.06 Vdc	0.01 Vdc		0.05 Vdc	
Rudder Operates Without Chatter or Instability	✓		✓	
CH 1 OPERATIONAL, CH 4 DISABLED				
Rudder Operates Without Chatter or Instability	✓		✓	
CH 4 OPERATIONAL, CH 1 DISABLED				
Rudder Operates Without Chatter or Instability	✓		✓	
MOMENTARILY SWITCHING LEFT RUDDER CAS OFF (SIMULATE COCKPIT SWITCH)				
Left Rudder Reverts to Trail Damp Mode	✓		✓	
Right Rudder Remains in CAS Mode	✓		✓	
After FCS Reset, Left Rudder is in CAS Mode	✓	✓	✓	✓
INTERFACE CONVERTER UNIT (ICU) RESET				
CHANNEL 1 RESET				
CH 4 Unaffected	✓		✓	
CH 1 Shut Off Time Expect: < 35 ms	31.6 ms		29.5 ms	
Transient During Shut Off Expect: 8% of full stroke	3.4% of full stroke		3.1% of full stroke	
Centertap Voltage Expect: 7.7 ± 0.3 Vrms	7.54 Vrms	7.56 Vrms	7.52 Vrms	7.52 Vrms
After FCS Reset, CH 1 Turns On to Control Actuator	✓		✓	
CHANNEL 4 RESET				
CH 1 Unaffected	✓		✓	
CH 4 Shut Off Time Expect: < 35 ms	31.8 ms		31.2 ms	
Transient During Shut Off Expect: 8% of full stroke	3.4% of full stroke		3.1% of full stroke	
Centertap Voltage Expect: 7.7 ± 0.3 Vrms	7.54 Vrms	7.56 Vrms	7.49 Vrms	7.49 Vrms
After FCS Reset, CH 4 Turns On to Control Actuator	✓		✓	
CHANNEL 1 MULTIPLE RESETS				
CH 1 Reset with FCS Reset Causes CH 1 to Turn On to Control Actuator. Can Be Repeated Multiple Times.	✓		✓	
CHANNEL 4 MULTIPLE RESETS				
CH 4 Reset with FCS Reset Causes CH 4 to Turn On to Control Actuator. Can Be Repeated Multiple Times.	✓		✓	
SIMULTANEOUS CH 1 & CH 4 MULTIPLE RESETS				
Left Rudder Reverts to Trail Damp Mode on Reset and is Restored to CAS Mode on FCS Reset. Can Be Repeated Multiple Times.	✓		✓	

TABLE 8. RUDDER INTEGRATED SYSTEM TEST SUMMARY (PART 3)

RUDDER INTEGRATED SYSTEM TEST SUMMARY (continued)	RUDDER SYSTEM (ICU, FRU, Actuator, and FCC for Channels 1 & 4)			
	SYSTEM 1		SYSTEM 2	
TEST	CHANNEL 1	CHANNEL 4	CHANNEL 1	CHANNEL 4
RUDDER SYSTEM RECOVERY FROM FAILURES				
Full CAS Operation is Recoverable Following a Single Channel Feedforward Command Fail and Recovery.	✓	✓	✓	✓
Full CAS Operation is Recoverable Following a Two Channel Feedforward Command Fail and Recovery.	✓		✓	
Full CAS Operation is Recoverable Following a Single Channel Feedback Position Fail and Recovery.	✓	✓	✓	✓
Full CAS Operation is Recoverable Following a Two Channel Feedback Position Fail and Recovery.	✓		✓ Channel 1 needed a 2nd system reset.	
RUDDER SYSTEM PERFORMANCE TESTS				
TRACKING BETWEEN CHANNELS				
Tracking Between CH 1 and CH 4 EHV Currents is <5% Full Scale.	✓		✓	
SENSOR FEEDBACK				
LVDT like Position Range is Within 5.376 Vrms (in and out of phase) Over ± 0.715 inches of Main Ram Travel.	✓		✓	
Position Scale Factor Expect: 7.519 ± 2% Vrms/inch	7.417 V/in	7.419 V/in	7.413 V/in	7.403 V/in
Maximum Deviation of Command vs Position Data Compared to Best Fit Line Through Data Expect: ≤25mVrms	-28.6mVrms @ 2.5" extend	22.6 mVrms @ 2.5" retract	24.5mVrms @ 2.5" extend	20.3 mVrms @ 2.5" extend
Maximum Difference Between CH1 & CH4 Expect: ≤0.05 Vrms	0.007 Vrms @ -10" retract		0.013 Vrms @ multiple commands	
Maximum Hysteresis Expect: ≤0.00143 inch	0.0010 inch		0.0010 inch	
ACTUATOR MOVEMENT THRESHOLD				
Input Voltage When Actuator Motion Starts Expect: < 5.38 mVrms	3.2 mVrms		1.7 mVrms	
Input Voltage When Actuator Motion Ceases Expect: < 5.38 mVrms	Motion still detected at 4.8 mVrms		Motion still detected at 2.1 mVrms	
FREQUENCY RESPONSE				
Small Amplitude Frequency Response, 1% full scale, with CH1 & Ch4 Operating Passes F/A-18 ATP Limits	✓		✓	
Small Amplitude Frequency Response, 1% full scale, with CH1 Operating, CH4 Disabled Passes F/A-18 A-D Acceptance Test Plan (ATP) Limits	✓		✓	
Small Amplitude Frequency Response, 1% full scale, with CH4 Operating, CH1 Disabled Passes F/A-18 ATP	✓		✓	
Large Amplitude Frequency Response, 10% full scale, with CH1 & Ch4 Operating Passes F/A-18 ATP Limits	✓		✓	
Large Amplitude Frequency Response, 10% full scale, with CH1 Operating, CH4 Disabled Passes F/A-18 A-D Acceptance Test Plan (ATP) Limits	✓		✓	
Large Amplitude Frequency Response, 10% full scale, with CH4 Operating, CH1 Disabled Passes F/A-18 ATP	✓		✓	

7.7.2 System Failure Modes and Effects Tests

The failure modes and effects tests inserted failures in each channel controlling the actuator and checked time to fail, fail detection, fail recovery, and actuator transients. A production fly-by-wire system was tested to obtain fail times that were used as expected fail times for the FACT system, although, greater fail times were allowed for some FACT monitors that added small amounts to FCC monitor fail times. The FACT fail monitors detect fails and send a fail signal to the FCC so the FCC fail monitors shut off the failed channel. The failure mode tests checked that the appropriate fail signal was sent to the FCC and the FCC shut off the failed channel. When the failure was removed and the system reset, recovery to normal operation was checked. Rudder actuator transients occurring from failures or failure recovery were checked against production specifications.

FACT failure monitors and FCC failure monitors covering the rudder actuator were tested by opening each electric and optic signal for actuator command or position. These signals were opened: the optic signal transmitting the FCC actuator command from ICU to FRU, the optic signal transmitting actual current through the actuator from FRU to ICU, the high and low signals for the FCC actuator command to the ICU, the high and low signals for the FRU actuator command to the actuator, the actuator position high, low, and centertap signals from ICU to FCC. The actuator spool LVDT position signal was opened to test an FCC actuator monitor not affected by the FACT system. The power connectors for the ICU and FRU were also opened to test the failure monitors.

Table 9 contains the system failure modes and effects test results. The test anomalies are discussed in the following paragraphs. See System Description section 6.1.2 for the System Monitors for an explanation of the FACT system failure monitors.

1. Rudder system one and two reversion times for the Single Channel Optic Failure created in the channel 4 actuator to ICU path exceeded the allowed 55 millisecond reversion time in the single channel optic failure section of the rudder failure modes and effects tests. The time to fail affects the aircraft transient during the fail. The time to fail was calculated from the one channel failure transient limit of 8% of main ram full stroke, 0.1144 inches (full stroke is 1.43 inches), and the no load rate for the rudder ram, 1.33 inches/second. 0.1144 / 1.33 = 86 ms or ~85 ms. The time allowed for the shut off valve to de-energize and the main ram to stop was measured during the FACT rudder Acceptance Test Procedure to be 30 ms. The remaining time, 55 ms, is left for the time from introduction of the failure to when the shut off valve diver no longer energizes the shut off valve. The 56 ms and 57 ms times to fail translate to a transient of 8.1% of full stroke. To make sure no problem exists, NASA-Dryden will retest the time to fail on the Iron Bird and verify the transient is acceptable.
2. Two test failures occurred in rudder system one in the Single Channel FCC/ICU Feedforward Electrical Interface Failure section of the rudder failure modes and effects test. The times to fail, reversion times, are about 17% longer for the integrated FACT and FCS system than for the production fly-by-wire system, but the reversion times are less than the longest reversion time for the production system, 89.8 milliseconds with the servovalve monitor causing the reversion. The cause for the longer times to fail is not known. The data from FACT system one varies more than FACT system two or the production system. The larger variance in times to fail could be due to the specific FCC amplifiers used in the separate tests. NASA-Dryden will test the failure times of the amplifier high and low side to compare results. It is expected that NASA-Dryden will measure different reversion times from the production system tested at Boeing due to the tolerances in the analog components from one FCC to another.

The integrated FACT and flight control system response to failures was excellent. Failures were always detected and reaction was quick. For the same fail, the FACT feedforward monitor along with the FCC centertap monitor reacted faster than the production system monitor, the FCC monitor with the FACT system reacted the same as the production system monitor, and the FACT feedback EOA software monitor along with the FCC centertap monitor reacted as expected. (i.e., about three times slower than the production system monitor due to the EOA software monitor detection time.)

TABLE 9. RUDDER SYSTEM FAILURE MODES AND EFFECTS SUMMARY (PART 1)

RUDDER FAILURE MODES AND EFFECTS						
FACT SYSTEM		SYSTEM #		PRODUCTION SYSTEM		COMPARISON
		1	2	(REFERENCE SYSTEM)		(FACT vs PRODUCTION)
TEST TITLE	PRIMARY FAILURE MONITOR	Revert Time (ms)	Revert Time (ms)	Revert Time (ms)	EQUIVALENT FAILURE in PRODUCTION CONFIGURATION	
SERVO VALVE MONITOR FAILURE (Production System Monitor Not Affected By FACT)						
Channel 1 Failed, Channel 4 Operational	FCC Servo Valve Monitor	65	65	89.8	Channel 1 spool position open on low side; apply full extend command	Differences attributed to differences in actuators
Channel 1 Failed, Channel 4 Operational	FCC Servo Valve Monitor	67	64	89.8	Channel 4 spool position open on high side; apply full extend command	Differences attributed to differences in actuators
SINGLE CHANNEL OPTIC FAILURE						
Failure in Channel 1 ICU to FRU	FACT Feedforward Monitor + FCC Centertap Monitor	42	49	50.2	Open Channel 1 EHV command high side; apply full extend command	FACT system equals or exceeds production system.
Failure in Channel 1 FRU to ICU	FACT Feedforward Monitor + FCC Centertap Monitor	45	51	61.2	Open Channel 1 EHV command low side; apply full extend command	FACT system equals or exceeds production system.
Failure in Channel 1 ICU to actuator	EOA Software + FCC Centertap Monitor	51	55	21.2	Open Channel 1 ram position high side	FACT performance is as expected: 15 bad decodes @ 2 msec/decode cycle + centertap @ 18 msec
Failure in Channel 1 actuator to ICU	EOA Software + FCC Centertap Monitor	51	54	17	Open Channel 1 ram position low side	FACT performance is as expected: 15 bad decodes @ 2 msec/decode cycle + centertap @ 18 msec
Failure in Channel 4 ICU to FRU	FACT Feedforward Monitor + FCC Centertap Monitor	46	44	50	Open Channel 4 EHV command high side; apply full extend command	FACT system equals or exceeds production system.
Failure in Channel 4 FRU to ICU	FACT Feedforward Monitor + FCC Centertap Monitor	54	50	62.2	Open Channel 4 EHV command low side; apply full extend command	FACT system equals or exceeds production system.
Failure in Channel 4 ICU to actuator	EOA Software + FCC Centertap Monitor	53	54	17.6	Open Channel 4 ram position high side	FACT performance is as expected: 15 bad decodes @ 2 msec/decode cycle + centertap @ 18 msec
Failure in Channel 4 actuator to ICU	EOA Software + FCC Centertap Monitor	56	57	17.2	Open Channel 4 ram position low side	FACT systems 1 and 2 reversion time higher than expected. Deserves re-test at Dryden.

TABLE 9. RUDDER SYSTEM FAILURE MODES AND EFFECTS SUMMARY (PART 2)

RUDDER FAILURE MODES AND EFFECTS (continued)						
FACT SYSTEM		SYSTEM #		PRODUCTION SYSTEM		COMPARISON
		1	2	(REFERENCE SYSTEM)		(FACT vs PRODUCTION)
SINGLE CHANNEL FCC / ICU FEEDFORWARD ELECTRICAL INTERFACE FAILURE						
Channel 1 Failed, amplifier high side	FCES Amplifier Monitor and/or FCES Spool Monitor	53	50	50.2	Open Channel 1 EHV command high side; apply full extend command	FACT system equivalent to production system within expected tolerance.
Channel 1 Failed, amplifier low side	FCES Amplifier Monitor and/or FCES Spool Monitor	72	58	61.2	Open Channel 1 EHV command low side; apply full extend command	FACT system 1 reversion time higher than production. Deserves re-test at Dryden.
Channel 4 Failed, amplifier high side	FCES Amplifier Monitor and/or FCES Spool Monitor	58	49	50	Open Channel 4 EHV command high side; apply full extend command	FACT system 1 reversion time higher than production. Deserves re-test at Dryden
Channel 4 Failed, amplifier low side	FCES Amplifier Monitor and/or FCES Spool Monitor	61	57	62.2	Open Channel 4 EHV command low side; apply full extend command	FACT system equivalent to production system within expected tolerance.
SINGLE CHANNEL ICU / FCC FEEDBACK ELECTRICAL INTERFACE FAILURE						
Channel 1 Failed, ram position high side	FCC Centertap Monitor	17	22	21.2	Open Channel 1 ram position high side	FACT system equivalent to production system within expected tolerance.
Channel 1 Failed, ram position low side	FCC Centertap Monitor	17	17	17	Open Channel 1 ram position low side	FACT system equivalent to production system within expected tolerance.
Channel 1 Failed, ram position center tap	FCC Centertap Monitor	15	14	14.6	Open Channel 1 ram position centertap	FACT system equivalent to production system within expected tolerance.
Channel 4 Failed, ram position high side	FCC Centertap Monitor	18	17	17.6	Open Channel 4 ram position high side	FACT system equivalent to production system within expected tolerance.
Channel 4 Failed, ram position low side	FCC Centertap Monitor	18	18	17.2	Open Channel 4 ram position low side	FACT system equivalent to production system within expected tolerance.
Channel 4 Failed, ram position center tap	FCC Centertap Monitor	15	14	14.4	Open Channel 4 ram position centertap	FACT system equivalent to production system within expected tolerance.

TABLE 9. RUDDER SYSTEM FAILURE MODES AND EFFECTS SUMMARY (PART 3)

RUDDER FAILURE MODES AND EFFECTS						
FACT SYSTEM		SYSTEM #		PRODUCTION SYSTEM		COMPARISON
		1	2	(REFERENCE SYSTEM)		(FACT vs PRODUCTION)
SINGLE CHANNEL FRU / ACTUATOR ELECTRICAL INTERFACE FAILURE						
Channel 1 Failed, rudder actuator command high side	FACT Feedforward Monitor + FCC Centertap Monitor	43	41	50.2	Open Channel 1 EHV command high side; apply full extend command	FACT system equals or exceeds production system.
Channel 1 Failed, rudder actuator command low side	FACT Feedforward Monitor + FCC Centertap Monitor	43	43	61.2	Open Channel 1 EHV command low side; apply full extend command	FACT system equals or exceeds production system.
Channel 4 Failed, rudder actuator command high side	FACT Feedforward Monitor + FCC Centertap Monitor	43	42	50	Open Channel 4 EHV command high side; apply full extend command	FACT system equals or exceeds production system.
Channel 4 Failed, rudder actuator command low side	FACT Feedforward Monitor + FCC Centertap Monitor	43	43	62.2	Open Channel 4 EHV command low side; apply full extend command	FACT system equals or exceeds production system.
LOSS OF FRU POWER						
Failure in Channel 1	FACT Feedforward Monitor + FCC Centertap Monitor	42	44	50.2	Open Channel 1 EHV command high side; apply full extend command	FACT system equals or exceeds production system.
Failure in Channel 4	FACT Feedforward Monitor + FCC Centertap Monitor	54	43	50	Open Channel 4 EHV command high side; apply full extend command	FACT system equivalent to production system within expected tolerance
LOSS OF ICU POWER						
Failure in Channel 1	FCC Centertap Monitor	16	15	14.6	Open Channel 1 ram position centertap	FACT system equivalent to production system within expected tolerance.
Failure in Channel 4	FCC Centertap Monitor	16	15	14.4	Open Channel 4 ram position centertap	FACT system equivalent to production system within expected tolerance.
TWO CHANNEL FAILURE AND RECOVERY						
The tests were performed with three different two channel fails		1) Channel 1 Feedforward Fail, Channel 4 Optic Fail		2) Channel 1 Optic Fail, Channel 4 Feedforward Fail		3) Channel 1 Servovalve Fail, Channel 4 Optic Fail
Left Rudder Reverts to Trail Damp Mode, Right Rudder Remains in CAS Mode		✓	✓	✓	Any two channel fail to cause the left rudder to revert to trail damp.	FACT system response is like the production system response.
Left Rudder is Restored to CAS Mode Following Removal of Fault, ICU Reset (if needed), and FCS Reset		✓	✓	✓	Remove fault and reset FCS.	FACT system response is like the production system response.

System tests were performed after component tests or airworthiness tests as the final system check before delivery to NASA-Dryden.

7.8 ICU and FRU Environmental Airworthiness Tests

Environmental airworthiness tests verified the ICU, FRU, and actuator can withstand the fighter aircraft environment for a 50 hour (one flight per week for one year) flight test program. Actuator airworthiness tests were mentioned in section 7.2, Actuator and Optic Sensor Acceptance and Airworthiness Tests. Airworthiness tests were performed on one ICU and one FRU, which qualified the other units by similarity, and consist of vibration, temperature and altitude, and electromagnetic compatibility tests. The ICU and FRU were monitored during these tests to verify they operated normally during environmental stress. The profiles of the tests correspond to the environment where each unit will be installed in the aircraft. (e.g., The FRU, located in the rear of the aircraft, has a more stressful vibration test than the ICU, located in the avionics bays near the cockpit.) A summary of the airworthiness test conditions for the ICU and FRU is shown in Table 10. After all airworthiness tests were complete, system tests verified the system operation.

TABLE 10. SUMMARY OF AIRWORTHINESS TEST CONDITIONS

Environmental Airworthiness Test	ICU and FRU Environmental Specifications
Altitude (Combined test with Temperature)	0 ft to 50,000 ft
Temperature (Combined test with Altitude)	$-40^{0}C$ to $85^{0}C$
Vibration: Sinusoidal Resonance Survey	Lesser of: $\pm 2g$ or 0.024 inch double amplitude
Vibration: Sinusoidal Resonance Dwell and Cycling Profile	0.2 inch to (0.055 inch for ICU) or (0.03 inch for FRU) displacement profile, two five minute dwells, 30 minutes cycling on profile
Vibration: Random	(4.5 g_{rms} for ICU) or (15.1 g_{rms} for FRU) for 30 min.
EMC: Radiated Emissions	MIL-STD-461C RE02 14kHz-10GHz
EMC: Radiated Susceptibility	MIL-STD-461C RS03 14kHz-10GHz at 200V/m
EMC: Conducted Emissions	MIL-STD-461C CE03 15kHz-50MHz, CE07 Spikes
EMC: Conducted Susceptibility	MIL-STD-461C CS01 30Hz-50kHz, CS02 50KHz-400MHz, CE06 Spikes

7.8.1 Temperature/Altitude Airworthiness Tests

Airworthiness temperature and altitude tests were performed on an ICU with the FRU at room temperature and on an FRU with the ICU at room temperature. The actuator was always at room temperature. The tests created temperature differences between the ICU and FRU to verify optic performance changes due to temperature do not create failures. There were three temperature changes at twelve to thirteen degrees per minute, and two, seventy minute temperature dwells at the extremes of $-40^{0}C$ and $85^{0}C$. During each temperature dwell, there were two altitude changes at 9880 feet per minute and a thirty minute altitude dwell at 50,000 feet. The target temperature/altitude profile is in Figure 38. The actual temperature for the ICU or FRU and the actual chamber temperature or pressure are in Figure 39 through Figure 42.

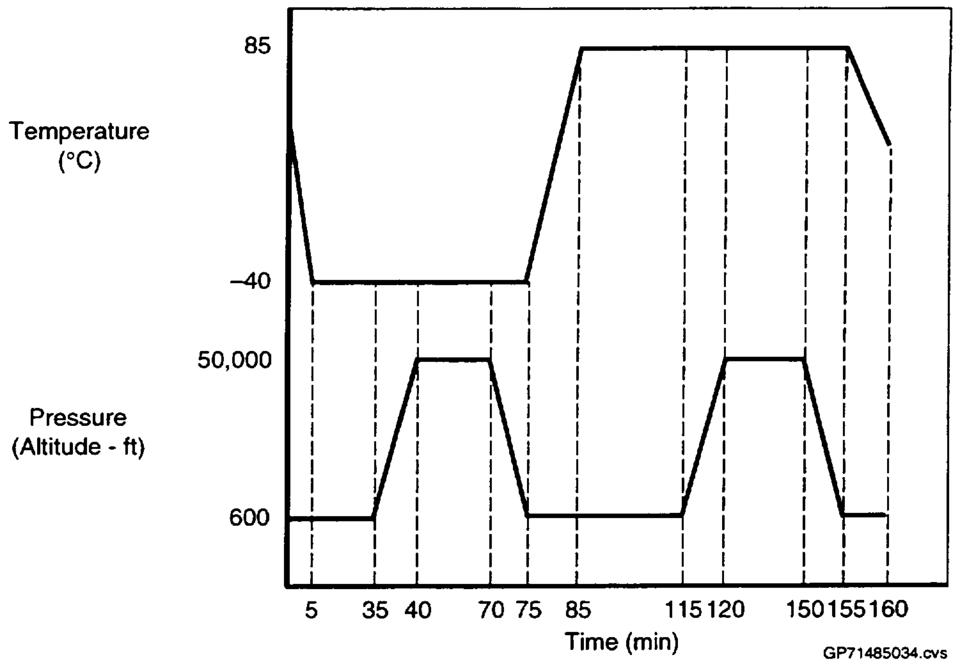


Figure 38. Temperature/Altitude Airworthiness Profile for ICU and FRU

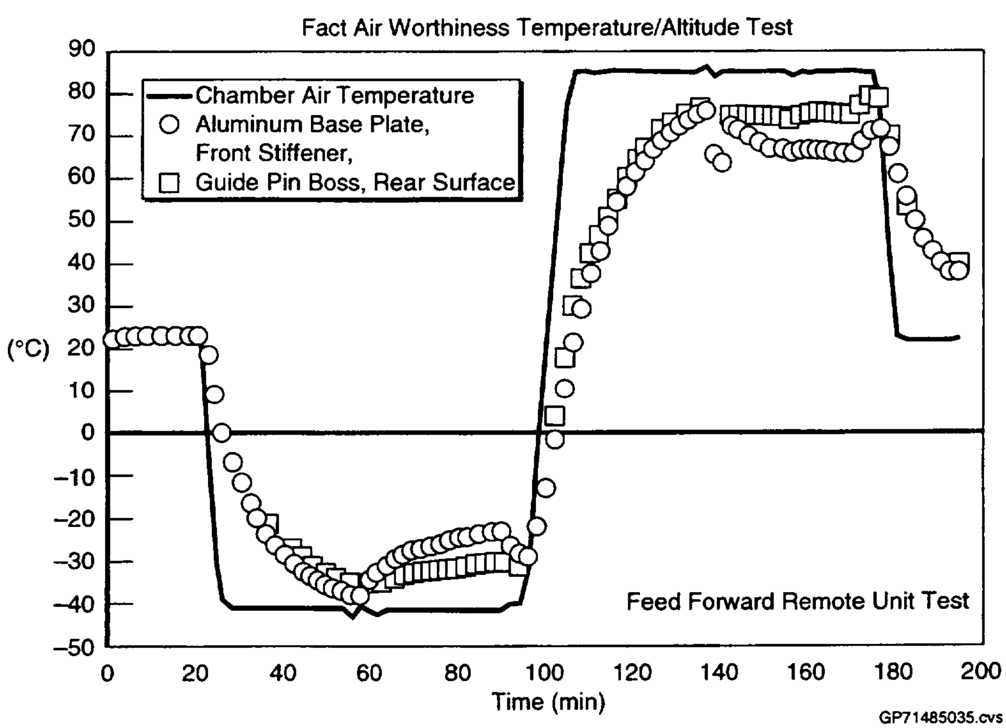


Figure 39. Actual Temperature for the Chamber and ICU

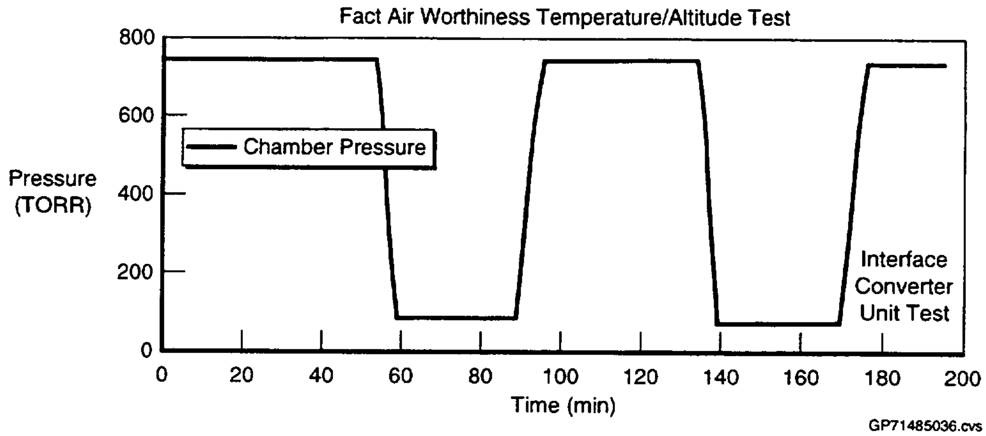


Figure 40. Actual Altitude Pressure for the Chamber During ICU Test

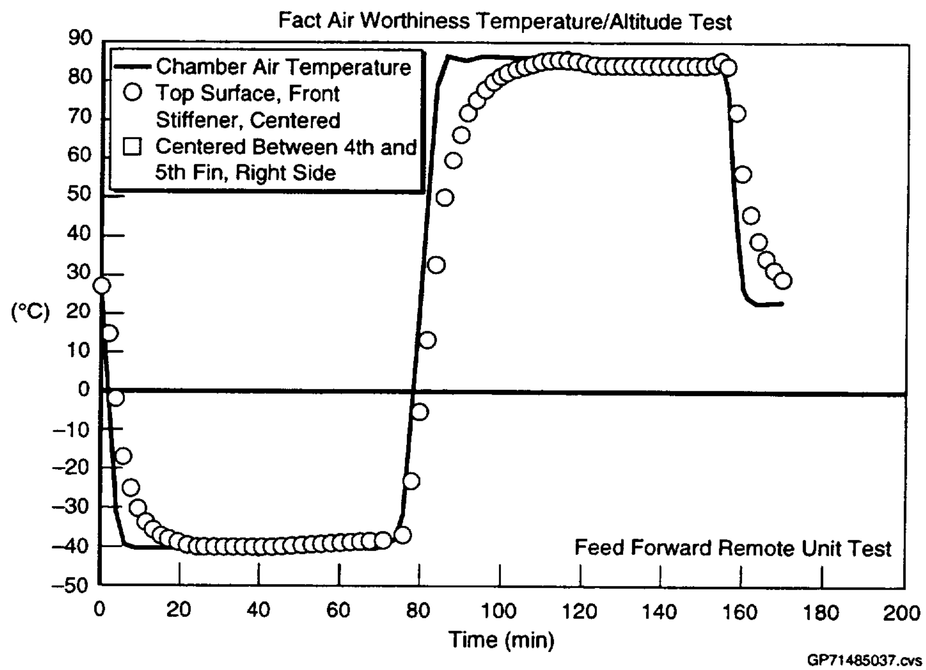


Figure 41. Actual Temperature for the Chamber and FRU

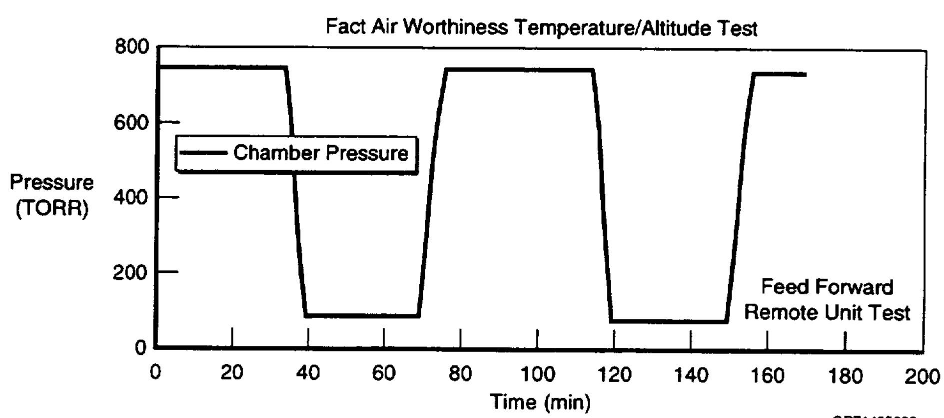


Figure 42. Actual Altitude Pressure for the Chamber During FRU Test

Table 11 contains the ICU and FRU temperature/altitude test results. There were no fails during the test, although, acceptable rudder decode errors occurred in the ICU test.

TABLE 11. TEMPERATURE/ALTITUDE AIRWORTHINESS TEST SUMMARY

TEMPERATURE / ALTITUDE AIRWORTHINESS TESTS								
System Status	ICU Current Command (volt)		FRU Current Command (volt)		Ram Position (Vrms)		LVDT Phase Shift (deg)	Integration Time (ms)
&	max.	min.	max.	min.	max.	min.	max&min	max&min
Comments	pretest ± 5%		pretest ± 5%		pretest ± 0.05		± 2 deg.	2 ms
No Fails, but decode errors and integration time increase during high temperature and high altitude dwell *	ICU in the Test Chamber, FRU at constant 22 °C							
pre-test:	-2.504		-2.494		0.1167 **		179.6	0.707
during test:	-2.504	-2.504	-2.494	-2.501	0.1167	0.1165	179.7 max 179.5 min	0.841 max 0.707 min
* – Rudder decode errors are acceptable, but indicate the EOA is having difficulty decoding the sensor. Decode errors are individual undecodeable sensor signals. Decode fails are at least 15 consecutive undecodeable sensor signals. The flight control system can operate with decode errors, but declares a failure and shuts off the channel after a rudder decode failure occurs. Decode errors are warning signs. ** – Ratio of position signal to reference signal instead of the position signal (by accident). Ratio varies ±0.0064 for a corresponding position variation of ±0.05. This assumes a constant reference.								
No Fails	FRU in the Test Chamber, ICU at constant 23 °C							
pre-test:	-2.504		-2.494		0.9050		179.5	0.841
during test:	-2.504	-2.505	-2.494	-2.499	0.9054	0.9048	179.5	0.841

During the ICU test, acceptable rudder decode errors occurred at 145 minutes, 153 minutes, and 155 minutes into the test. All errors occurred during the high temperature and high altitude dwell (85^0C and 50,000 feet). Decoding errors are acceptable until they occur fifteen consecutive times and cause a decode fail. Along with the decode errors, the EOA actual and estimated integration time increased from 0.707 milliseconds to 0.841 milliseconds. The integration time returned to 0.707 milliseconds after the altitude had reduced to St. Louis altitude and the ICU chassis temperature had dropped to 48^0C.

At the high temperature and altitude, the EOA needed more time to gather sensor light and was unable to decode some sensor signals. While no failures occurred and the EOA integration time was less than half of the two millisecond maximum, the decode errors and increased integration time warn that the EOA had increased difficulty decoding the sensor. The difference in temperature between the EOA and sensor does not matter since the optic transmitter and receiver are in the EOA and the sensor is passive.

The position feedback system passed with a few decode errors but no failures, the command feedforward system passed without errors, and neither the ICU nor FRU was affected by having the other unit at different temperatures. The ICU and FRU performed well over the temperature and altitude range.

7.8.2 Vibration Airworthiness Tests

Airworthiness vibration testing consisted of three different vibration tests performed in each of three axes on an ICU and on an FRU. The sinusoidal resonance survey test was used to find chassis vibration resonances up to 2000 hertz. The sinusoidal cycling test subjected the chassis to thirty minutes of sinusoidal vibration over a low frequency range, 5 to 50 hertz for the ICU and 5 to 85 hertz for the FRU. Sinusoidal dwell tests would have subjected the ICU and FRU to five minute dwells at the two most severe resonances in the cycling test frequency range but were not performed since there were no resonances in that range. The random vibration test subjected the chassis to thirty minutes of random vibration, 4.5 g_{rms}^2/hertz for the ICU and 15.1 g_{rms}^2/hertz for the FRU.

7.8.2.1 Resonance Survey

The profile for the resonance survey was a ten minute sinusoidal sweep from 5 hertz to 2000 hertz of the lesser of 0.024 inch double amplitude displacement or ±2g.

Table 12 contains the resonance survey vibration test results. There were no fails during the test.

TABLE 12. VIBRATION RESONANCE SURVEY SUMMARY

VIBRATION AIRWORTHINESS RESONANCE SURVEY TESTS								
System Status	ICU Current Command (volt dc)		FRU Current Command (volt dc)		Ram Position (volt rms)		LVDT Phase Shift (degrees)	
	pre-test	post-test	pre-test	post-test	pre-test	post-test	pre-test	post-test
	LIMITS:	pretest ±5%		pretest ±5%		pretest ±0.05	± 2 deg	± 2 deg
No Fails	ICU in the VERTICAL AXIS							
	1st Resonant Frequency (Hertz):			350	Transmissibility:		2.5	
	2nd Resonant Frequency (Hertz):			450	Transmissibility:		2.5	
	-2.507	-2.506	-2.499	-2.500	0.964	0.962	178.4	178.4
No Fails	ICU in the LONGITUDINAL AXIS							
	1st Resonant Frequency (Hertz):			224	Transmissibility:		2.7	
	2nd Resonant Frequency (Hertz):			256	Transmissibility:		4.1	
	-2.506	-2.506	-2.500	-2.502	1.003	1.026	178.6	178.6
No Fails	ICU in the LATERAL AXIS							
	1st Resonant Frequency (Hertz):			144	Transmissibility:		3.8	
	2nd Resonant Frequency (Hertz):			221	Transmissibility:		2.8	
	-2.507	-2.507	-2.499	-2.499	1.004	1.004	178.6	178.6
No Fails	FRU in the VERTICAL AXIS							
	1st Resonant Frequency (Hertz):			1076	Transmissibility:		2.5	
	2nd Resonant Frequency (Hertz):			1213	Transmissibility:		2.5	
	-2.507	-2.507	-2.499	-2.499	0.963	0.964	178.6	178.6
No Fails	FRU in the LONGITUDINAL AXIS							
	1st Resonant Frequency (Hertz):			1120	Transmissibility:		2.5	
	2nd Resonant Frequency (Hertz):			1648	Transmissibility:		5.3	
	-2.506	-2.506	-2.502	-2.502	0.963	0.963	178.6	178.6
No Fails	FRU in the LATERAL AXIS							
	1st Resonant Frequency (Hertz):			632	Transmissibility:		2	
	2nd Resonant Frequency (Hertz):			1076	Transmissibility:		2.6	
	-2.506	-2.507	-2.499	-2.499	1.000	1.002	178.6	178.7

The survey results were very good for the ICU and FRU. All resonances were beyond the frequency range of the sinusoidal cycling profiles so no dwell tests were required. The transmissibilities were fairly low. Most were near 2.5, the ICU maximum was 4.1, and the FRU maximum was 5.3.

7.8.2.2 Sinusoidal Cycling

The profiles for the ICU and FRU sinusoidal cycling are in Figure 43 and Figure 44 respectively. The performance profiles were used.

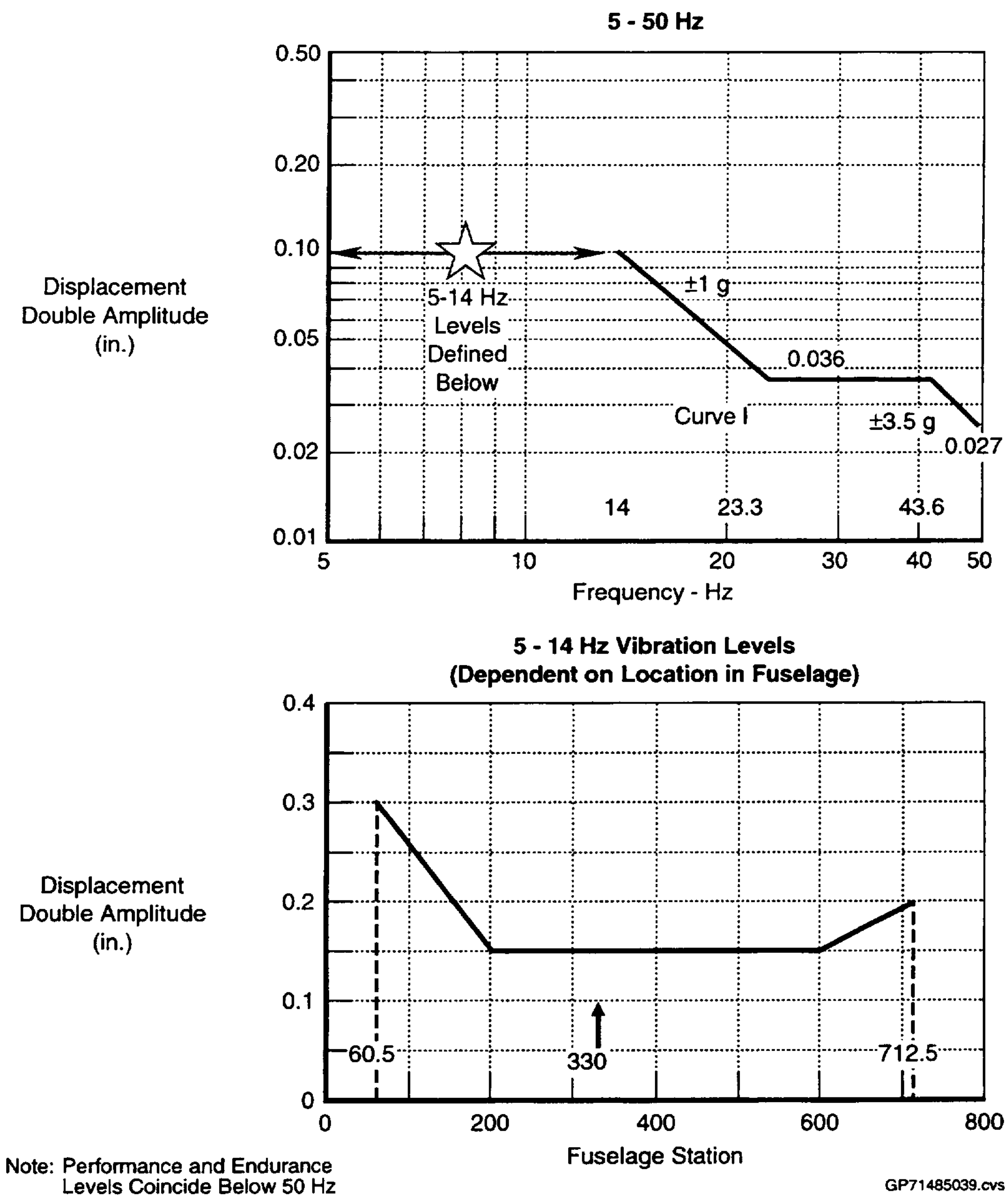


Figure 43. Sinusoidal Cycling Vibration for ICU

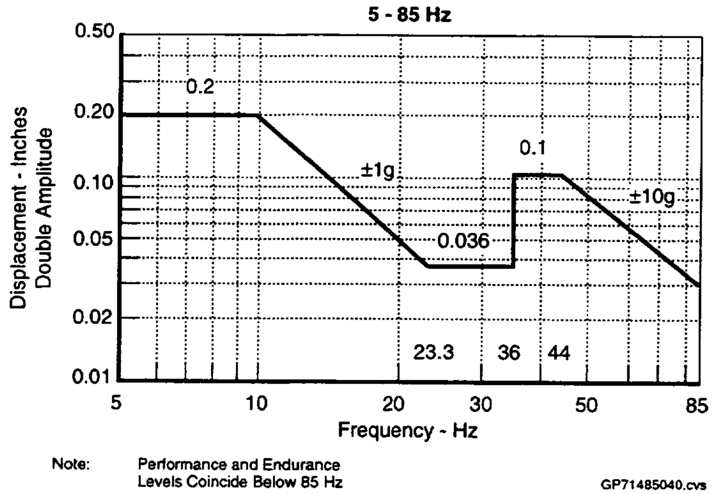


Figure 44. Sinusoidal Cycling Vibration for FRU

Table 13 contains the sinusoidal cycling vibration test results. There were no fails during the test.

TABLE 13. VIBRATION SINUSOIDAL CYCLING SUMMARY

VIBRATION AIRWORTHINESS SINUSOIDAL CYCLING TESTS								
System Status	ICU Current Command (volt)		FRU Current Command (volt)		Ram Position (Vrms)		LVDT Phase Shift (deg)	Integration Time (ms)
&	max.	min.	max.	min.	max.	min.	max&min	max&min
Comments	pretest ± 5%		pretest ± 5%		pretest ± 0.05		± 2 deg.	≤2 ms
No Fails	ICU in the VERTICAL AXIS							
pre-test:	-2.507		-2.501		0.992		178.6	0.841
during test:	-2.506	-2.507	-2.497	-2.503	0.968	0.963	178.6	0.841
No Fails	ICU in the LONGITUDINAL AXIS							
pre-test:	-2.506		-2.502		1.013		178.6	0.841
during test:	-2.506	-2.507	-2.498	-2.500	1.013	1.009	178.6	0.841
No Fails	ICU in the LATERAL AXIS							
pre-test:	-2.506		-2.502		1.002		178.6	1.000
during test:	-2.506	-2.507	-2.498	-2.503	1.005	1.001	178.6	1.000
No Fails	FRU in the VERTICAL AXIS							
pre-test:	-2.507		-2.499		0.964		178.6	0.841
during test:	-2.506	-2.507	-2.498	-2.500	0.964	0.963	178.6	0.841
No Fails	FRU in the LONGITUDINAL AXIS							
pre-test:	-2.506		-2.502		0.963		178.6	0.841
during test:	-2.506	-2.507	-2.499	-2.503	0.963	0.963	178.6	0.841
No Fails	FRU in the LATERAL AXIS							
pre-test:	-2.507		-2.499		1.000		178.6	1.000
during test:	-2.506	-2.507	-2.498	-2.499	1.002	1.000	178.6	1.000

All test results were well within the tolerances. There were no affects due to sinusoidal cycling.

7.8.2.3 Random

The profiles for ICU and FRU random vibration are in 45 and 46. The performance profiles were used.

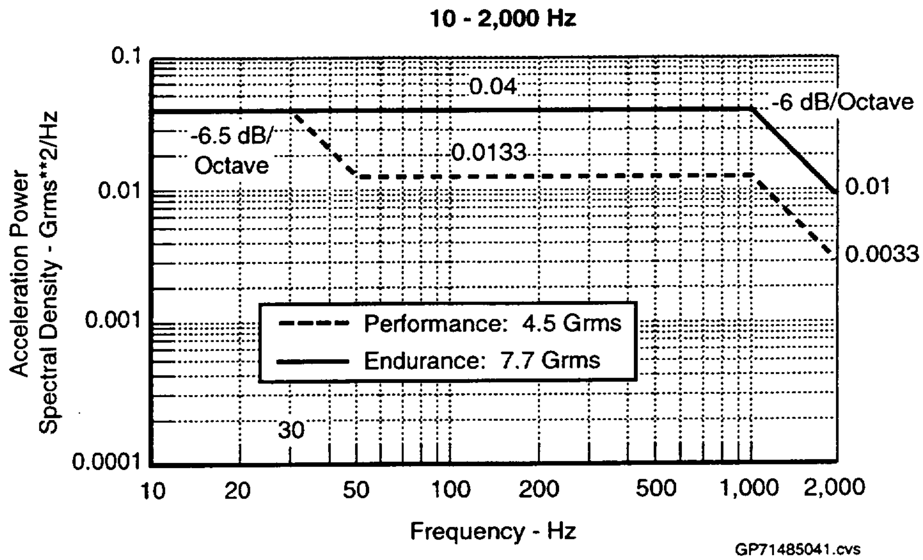


Figure 45. Random Vibration for ICU

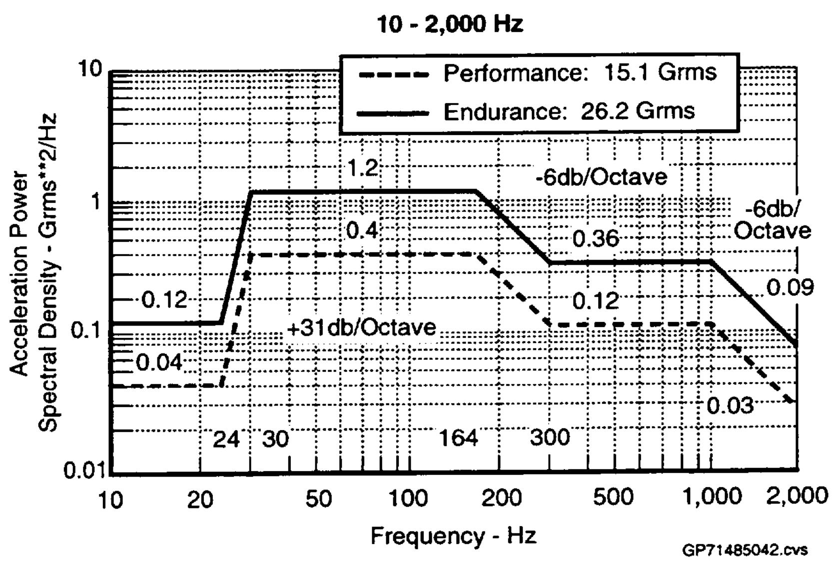


Figure 46. Random Vibration for FRU

Table 14 contains the random vibration test results. There were no fails during the test.

TABLE 14. VIBRATION RANDOM SUMMARY

VIBRATION AIRWORTHINESS RANDOM TESTS								
System Status	ICU Current Command (volt)		FRU Current Command (volt)		Ram Position (Vrms)		LVDT Phase Shift (deg)	Integration Time (ms)
&	max.	min.	max.	min.	max.	min.	max&min	max&min
Comments	pretest ± 5%		pretest ± 5%		pretest ± 0.05		± 2 deg.	≤2 ms
No Fails	ICU in the VERTICAL AXIS							
pre-test:	-2.506		-2.499		0.963		178.6	0.841
during test:	-2.506	-2.507	-2.498	-2.500	0.964	0.963	178.6	0.841
No Fails	ICU in the LONGITUDINAL AXIS							
pre-test:	-2.506		-2.499		1.008		178.6	0.841
during test:	-2.506	-2.507	-2.498	-2.500	1.008	1.002	178.6	0.841
No Fails	ICU in the LATERAL AXIS							
pre-test:	-2.506		-2.499		1.004		178.6	1.000
during test:	-2.506	-2.507	-2.498	-2.500	1.005	1.001	178.6	1.000
No Fails	FRU in the VERTICAL AXIS							
pre-test:	-2.506		-2.499		0.963		178.6	0.841
during test:	-2.506	-2.507	-2.498	-2.500	0.964	0.963	178.6	0.841
No Fails	FRU in the LONGITUDINAL AXIS							
pre-test:	-2.506		-2.499		0.963		178.6	0.841
during test:	-2.506	-2.507	-2.499	-2.503	0.963	0.962	178.6	0.841
No Fails	FRU in the LATERAL AXIS							
pre-test:	-2.506		-2.499		1.001		178.6	1.000
during test:	-2.506	-2.507	-2.498	-2.500	1.001	0.999	178.6	1.000

All test results were well within the tolerances. There were no affects due to random vibration.

The ICU and FRU performed perfectly in the airworthiness vibration tests. Resonances were at relatively high frequencies for avionics chassis and system performance was flawless.

7.8.3 Electromagnetic Compatibility Airworthiness Tests

Airworthiness electromagnetic compatibility (EMC) tests were performed on an ICU and FRU together to provide insight as to how the FACT system will affect other aircraft systems and how other aircraft systems will affect the FACT system. NASA-Dryden will conduct a hanger radiation test that will determine the FACT system's electromagnetic compatibility with the aircraft. MIL-STD-461C limits, MIL-STD-462C procedures, and Boeing EMC Facility best practices were used to gather the data and provide target performance. Radiated emissions test RE02 measured the electric field emissions radiating from the chassis. Conducted emissions tests CE03 and CE07 measured the emissions conducted on the wires carrying power to the system. Conducted susceptibility tests CS01, CS02, and CS06 determined the ability of the system to survive noise on the power lines. Radiated susceptibility test RS03 determined the ability of the system to survive in an electric field.

The FACT system did not meet all MIL-STD-461 C Notice 2 requirements for CE03, RE02, and RS03 but did successfully meet the requirements for CE07, CS01, CS02, and CS06. Refer to the actual EMC test report for more detailed information than is presented in this summary.

7.8.3.1 EMC Facility Description

The FACT system was tested in the Boeing EMC Facility's shielded anechoic chamber. The dimensions of the chamber are 20 feet by 24 feet by 12 feet, and the walls and ceiling are covered with 24 inch pyramidal RF absorber. The ground plane is a solid copper sheet measuring eleven by four feet. The FACT support test equipment was located in an adjacent shielded room measuring sixteen by sixteen feet. The 28 VDC power to the shielded room and anechoic chamber are filtered by wall mounted filters, and the 28 VDC power to the FACT system was supplied from the anechoic chamber. The FACT cable access into the chamber was through an access panel in the chamber wall.

7.8.3.2 EMC Test Pass/Fail Criteria

Pass/fail criteria are different for emissions and susceptibility tests. Pass/fail criteria for the emissions tests CE03, CE07, and RE02 are directly given in MIL STD 461C as emissions limits. The FACT system passed or failed these tests based upon whether or not the measured emissions from the unit under test (UUT) exceeded the limits in the military standard. Failure criteria for the susceptibility tests CS01, CS02, CS06, and RS03 are any anomaly, degradation of performance, or change in the proper operation of the FACT system. Using support test equipment, system performance was determined by monitoring four discrete parameters from the FACT system: ICU command current, FRU command current, modulated LVDT actuator position, and the phase shift of the modulated LVDT output. Use of the support equipment computer to monitor test points from the FACT system was limited due to the susceptibility of the computer to the test conditions required.

7.8.3.3 EMC Test Descriptions

The FACT system was tested to the limits and requirements specified by MIL-STD-461C Notice 2 for class A1b equipment. The airworthiness tests were conducted in general accordance with MIL-STD-462 Notice 6 and the Boeing EMC Facility's standard practices given the cabling and support equipment provided. Specific test parameters (e.g., resolution bandwidths, minimum sweep times, RF modulations, etc.) were not specified so the generic parameters listed in Table 15, Table 16, and Table 17 were used.

Table 15. RS03 RF MODULATION REQUIREMENTS

Start Frequency	Stop Frequency	Modulation Used
14 kHz	400 MHz	80 % AM, 400 Hz sinusoid
400 MHz	1000 MHz	100 % AM, 1 kHz square wave, 50 % duty cycle
1 GHz	10 GHz	5 us width, 1 kHz Pulse

Table 16. RE02 BANDWIDTHS

Frequency Range	Resolution Bandwidth Filter Size
14 – 500 kHz	1 kHz
0.5 – 25 MHz	3 kHz
25 – 60 MHz	10 kHz
60 – 150 MHz	30 kHz
150 – 700 MHz	100 kHz
700 – 1000 MHz	30 kHz
1 – 10 GHz	10 kHz

TABLE 17. CE03 BANDWIDTHS

Frequency Range	Resolution Bandwidth Filter Size
15 – 100 kHz	3 kHz
100 – 700 kHz	10 kHz
0.7 – 50 MHz	30 kHz

7.8.3.3.1 Nonconformities To Standard Test Setup And Procedures

The FACT test setup had several deviations from the typical test setup used by the Boeing EMC Test Facility which may have adversely affected the final test results. As stated previously, the FACT EMC Airworthiness Tests were conducted in general accordance with MIL STD 462 Notice 6 and the standard practices of the Boeing EMC Test Facility, however, the EMC airworthiness tests were conducted on a very informal basis. This resulted in several irregularities in the EMC test setup and test methodologies that would not have been present in a more structured test setup. The end result of these irregularities is that the EMC test results may have been somewhat compromised in areas of repeatability, accuracy of the frequency and amplitude quantitative measurements, and the similarities to the performance of the system as installed in the aircraft. The irregularities are listed below.

1. A suitable mounting fixture was not available to simulate the aircraft method of electrical bonding of the ICU to the aircraft structure. The method used consisted of placing strips of copper from the bonding area on the rear of the ICU to the copper ground plane. The mechanical method to hold the copper strips to the ICU chassis was not ideal, and the electrical bond tended to degrade over time. The bonding of the ICU was reseated and rechecked at the start of each EMC test to ensure the value was below the typical 2.5 milliohm requirement.
2. The electrical cabling provided was the cabling used for the FACT laboratory integration testing and was not physically representative of the cable layout or wire to wire routing of the aircraft wiring since stabilator wiring was included that the aircraft will not use. The cabling, as arranged on the ground plane, had several wires that exited the ICU, ran the length of the two meters of required test cabling, and then physically looped back on themselves along the same two meters of cabling back to the FRU. The loops were minimized as much as possible but could not be eliminated. In addition, there were numerous wires and unterminated connectors present in the cable bundles which were not used for the EMC testing but were part of the cable bundle provided and were present on the ground plane throughout the emissions and susceptibility testing. Several of these wires that entered the ICU or FRU but were not terminated at the opposite end were removed from the ICU or FRU prior to testing. The types of wire (e.g., shielded twisted pair) used in the FACT system test cables were the same as the types of wire suggested for use in the test aircraft. The electrical design of these cables (i.e., backshells, shield terminations, wire types, etc...), may have contributed significantly to the poor RS03 performance of the FACT system.
3. The cabling to the FACT support equipment in the shielded anteroom was of minimal EMC design and consisted of discrete wires and ribbon cable that were not shielded. The wiring was shielded as a whole, not individual wires, by placing aluminum foil over the wiring and copper taping it to the copper ground plane. This may have contributed to wire to wire coupling and the possible susceptibilities of the test computer.
4. Near the end of EMC testing, it was discovered that two FACT system wires were missing from the test cabling. These two wires were the 28 VDC Battery Backup and 28 VDC Battery Backup Return. These two wires would have normally entered the ICU and were not present during the radiated emissions and susceptibility testing and were not tested for possible FACT system susceptibilities during the conducted susceptibility tests. No EMC tests were repeated with the Backup 28 VDC wires in place.

7.8.3.4 EMC Test Results

7.8.3.4.1 CE03, Conducted Emissions, Power Lines, 15 kHz – 50 MHz

Conducted emissions testing was performed on the primary 28 VDC power line to the FACT ICU. Minor outages above the test limits were noted from approximately 1 – 2 MHz (broadband emissions) and sporadically between 16 and 23 MHz (narrowband emissions). The worst case outage was 5.6 dB over the test limit. The 28 VDC Return line was not tested because it is grounded internally to the ICU and therefore does not meet the applicability of CE03. The 28 VDC Battery Backup power line was not tested.

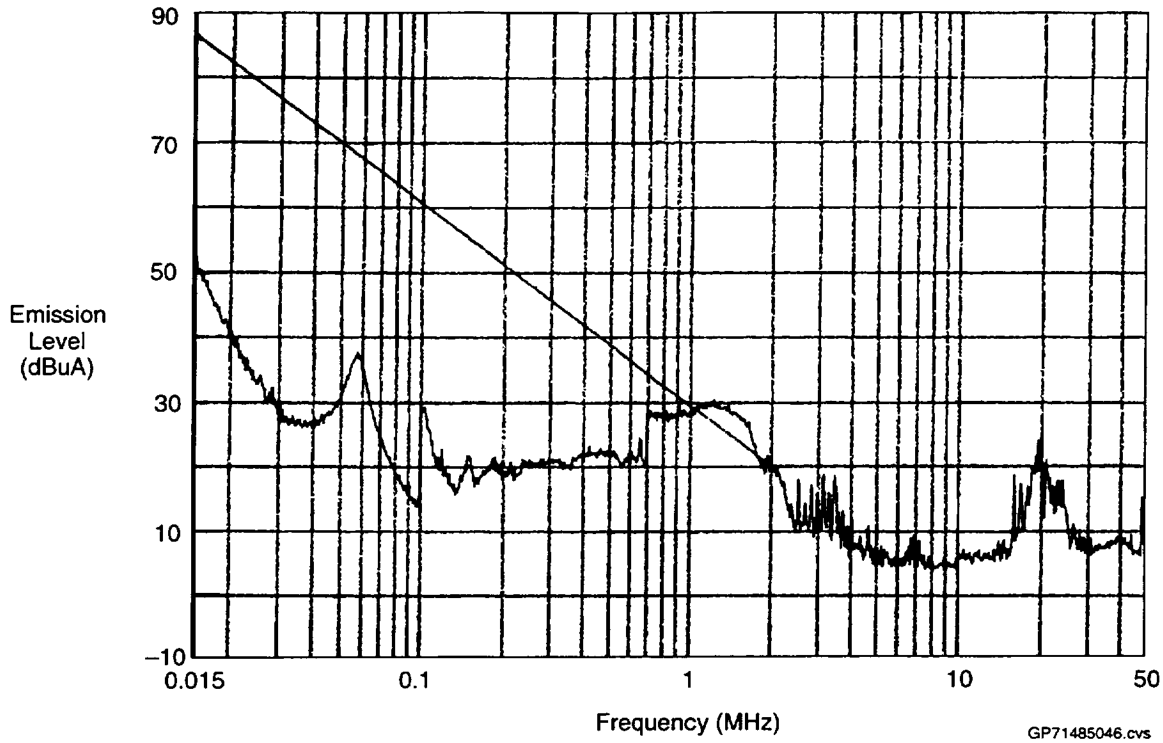


Figure 47. CE03 Narrowband Conducted Emission

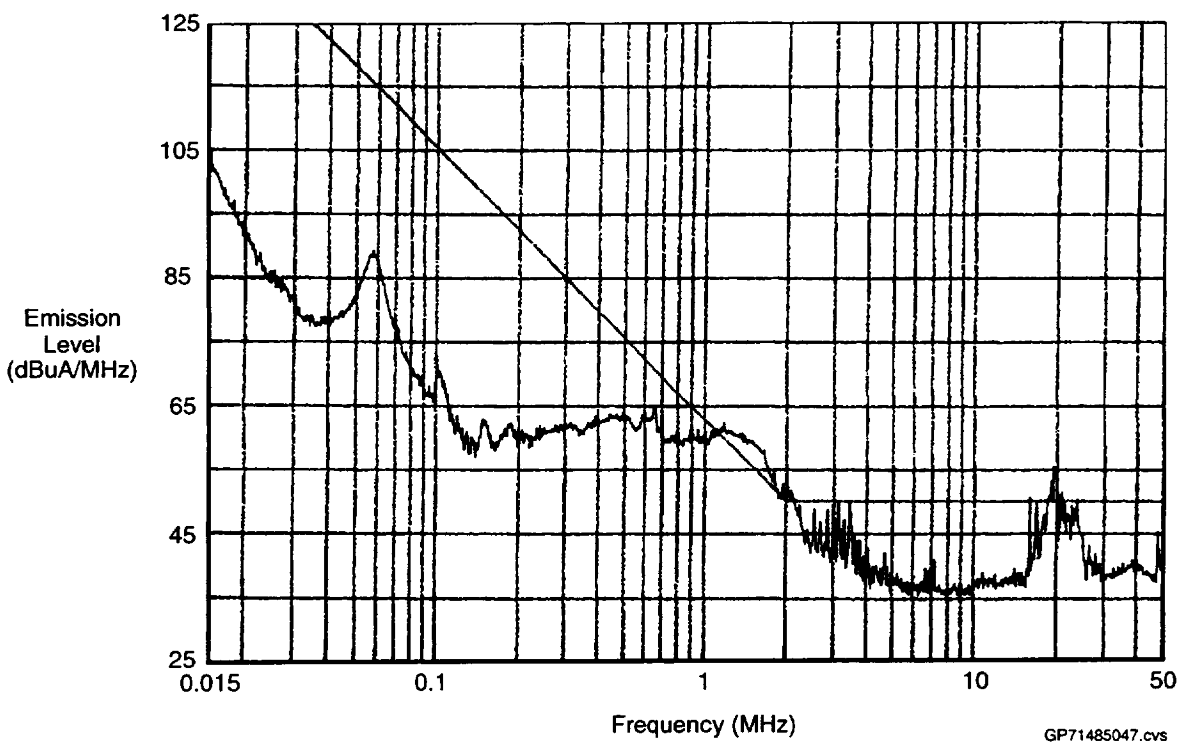


Figure 48. CE03 Broadband Conducted Emission

7.8.3.4.2 CE07, Conducted Emissions, Power Leads, Spikes, Time Domain

Conducted emissions testing was performed on the primary 28 VDC power line to the FACT ICU. No FACT power transients were detected which exceeded the test limits so the system meets the requirements of CE07. The voltage spiked to a maximum 31 VDC and a minimum +11 VDC. The test limits for CE07 state that transients of less than 50 microseconds duration shall not exceed + 50 % (28 V + 14 Volts) or - 150 % (28 V – 42 V) of the nominal DC line voltage of 28 V. The FACT was monitored for transients as the unit was powered on and as the unit was powered off. The 28 VDC Return line was not tested because it is grounded internally to the ICU. The 28 VDC Battery Backup power line was not tested.

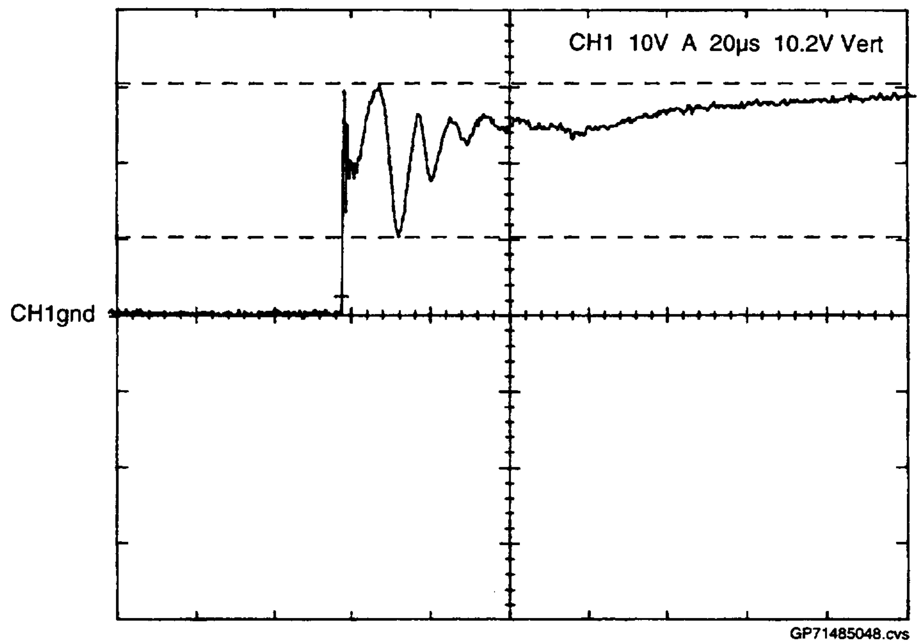


Figure 49. CE07 Power On Transient Oscilloscope Display

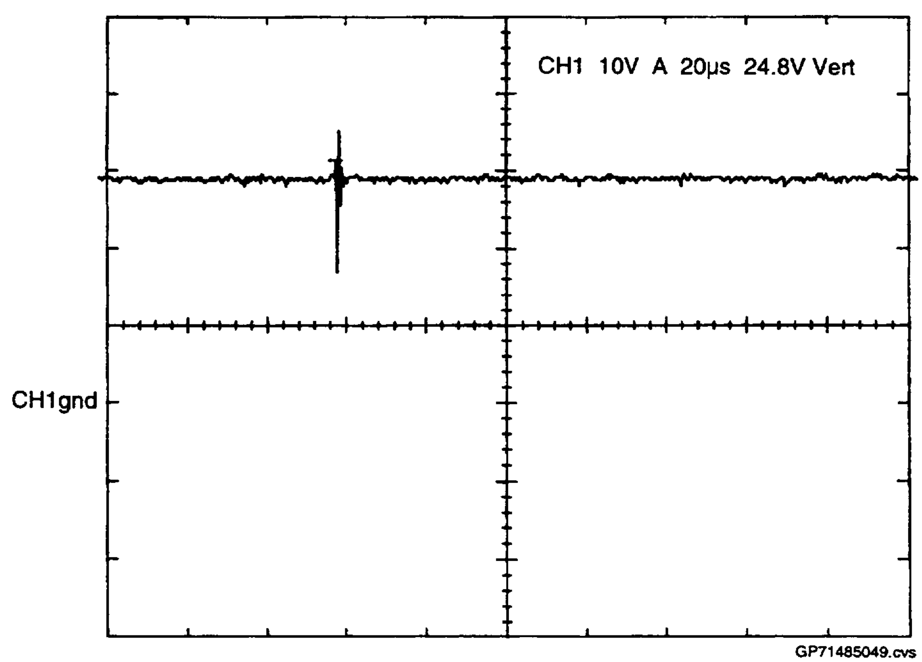


Figure 50. CE07 Power Off Transient Oscilloscope Display

7.8.3.4.3 CS01, Conducted Susceptibility, Power Leads, 30 Hz – 50 kHz

Conducted susceptibility testing was performed on the primary 28 VDC power line to the FACT ICU. No FACT susceptibilities were noted during this test so the system meets the requirements of CS01. The test limits for CS01 are given in MIL STD 461C Notice 2 and vary across the frequency band from 2.8 Vrms down to 1.0 Vrms. The 28 volt test limit is in Figure 51. The standard also states the test requirement will be considered met if the required audio amplifier, adjusted to dissipate 50 Watts into a 0.5 Ohm load, can not generate the required voltage on the power line at that frequency. The FACT 28 VDC power line met one of these two conditions across the frequency band. The 28 VDC Return line was not tested because it is grounded internally to the ICU and therefore does not meet the applicability of CS01. The 28 VDC Battery Backup power line was not tested.

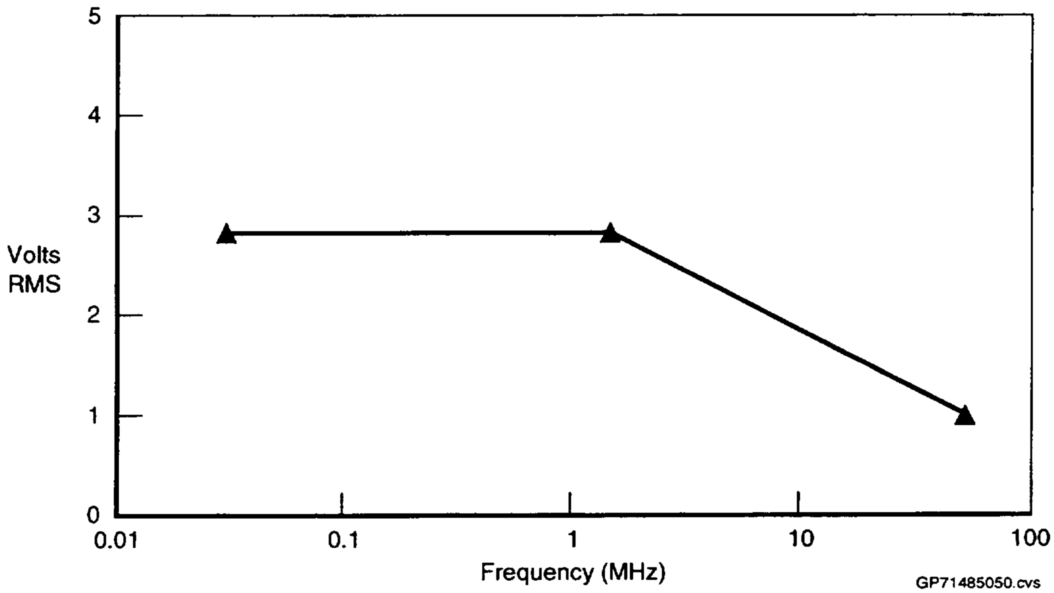


Figure 51. CS01 28 Volt Test Limit

7.8.3.4.4 CS02, Conducted Susceptibility, Power Leads, 50 kHz – 400 MHz

Conducted susceptibility testing was performed on the primary 28 VDC power line to the FACT ICU. No FACT susceptibilities were noted during this test so the system meets the requirements of CS02. The test limit for CS02 is 1 Vrms injected at the connector of the UUT. The military standard also states that the UUT will also be considered to have met the test requirements when a 1 Watt source of 50 ohms impedance cannot develop the required voltage at the test sample input terminals, and the UUT is not susceptible to the output of the signal source. The FACT system successfully met one of these two requirements across the frequency band. The 28 VDC Return line was not tested because it is grounded internally to the ICU and therefore does not meet the applicability of CS02. The 28 VDC Battery Backup power line was not tested.

7.8.3.4.5 CS06, Conducted Susceptibility, Spikes, Power Leads

Conducted susceptibility testing was performed on the primary 28 VDC power line to the FACT ICU. No FACT susceptibilities were noted during this test so the system meets the requirements of CS06. The test requirement for CS06 is the unit must be immune to injected spikes of pulse durations of 0.15 and 10.0 microseconds of both positive and negative 200 Vpeak polarities. The 28 VDC Return line was not tested because it is grounded internally to the ICU and therefore does not meet the applicability of CS02. The 28 VDC Battery Backup power line was not tested.

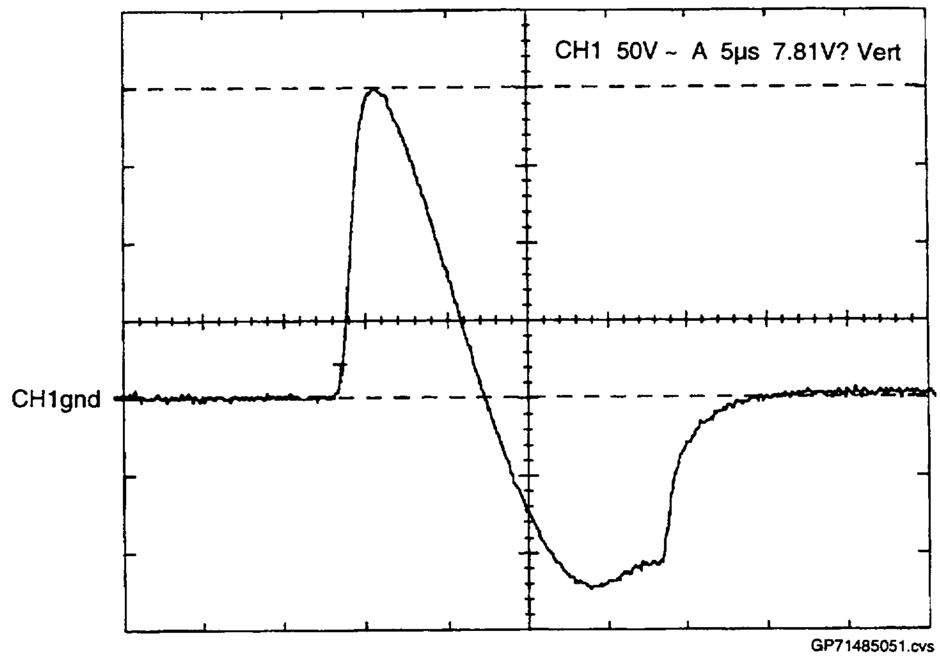


Figure 52. CS06 Oscilloscope Display of Transient from 10µs Width 200 Volt Spike

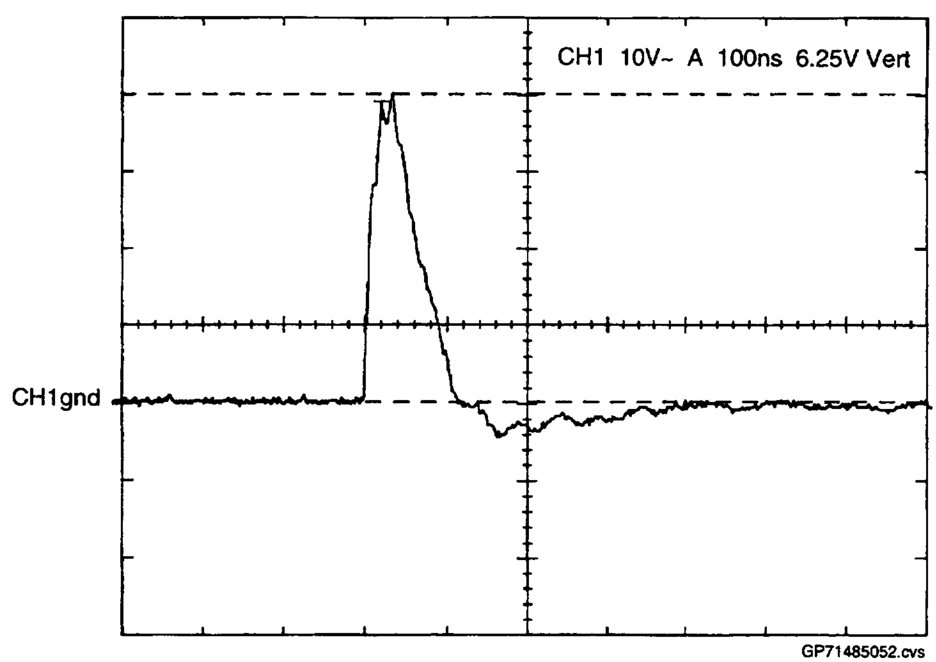


Figure 53. CS06 Oscilloscope Display of Transient from 0.15µs Width 200 Volt Spike

7.8.3.4.6 RE02, Radiated Emissions, Electric Field, 14 kHz – 10 GHz

Radiated emissions testing was performed on the FACT system from 14 kHz to 10 GHz. Measurements were made in both vertical and horizontal antenna polarizations in the applicable frequency bands. Outages above the test limit were recorded at frequencies as low as 1 MHz and up to 145 MHz. The worst case outage was 39.2 dB over the test limit at 16.06 MHz. The measurement scans of the test chamber and support equipment ambients (scans made with all support equipment on but FACT power off) had a few frequencies which were above the ambient limits of six decibels below the test limit and had a frequency, 14.3 MHz, in the vertical polarization where the ambient limits exceeded the test limit by 6 dB for narrowband and 10 dB for broadband emissions. All those emissions were deemed acceptable. The effects of the 28 VDC Battery Backup power lines were not tested.

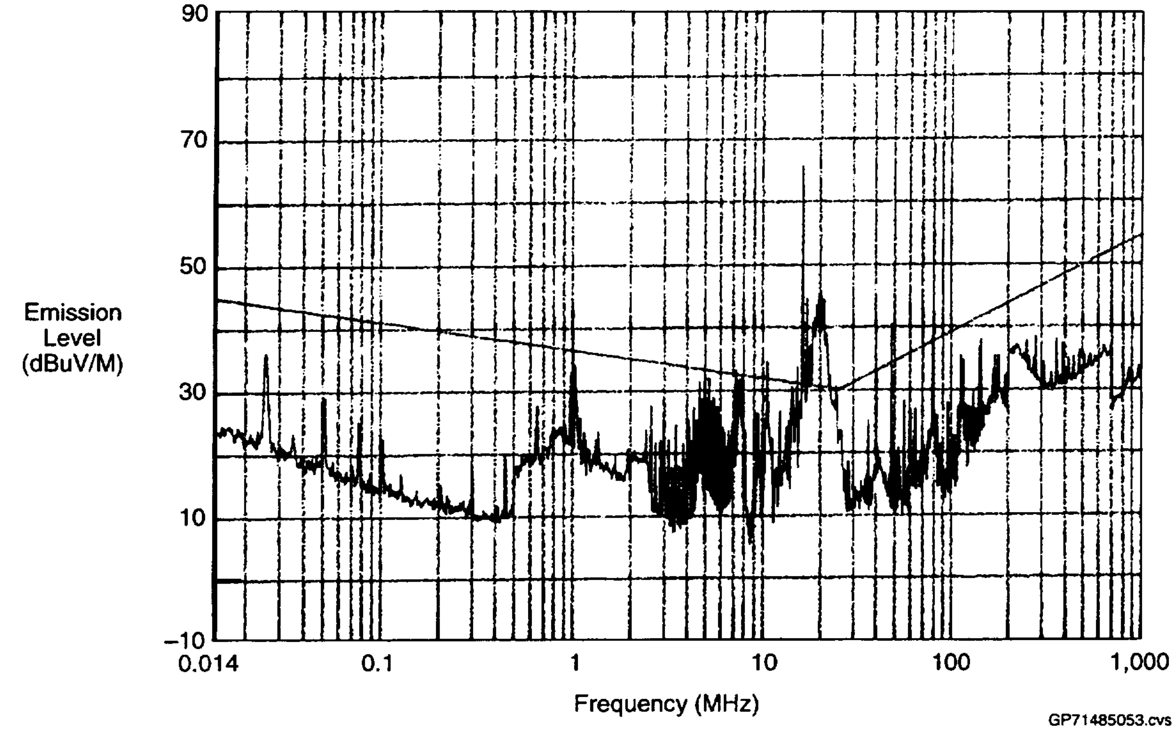


Figure 54. RE02 Horizontally Polarized 0.014 to 1000 MHz Narrowband Radiated Emission

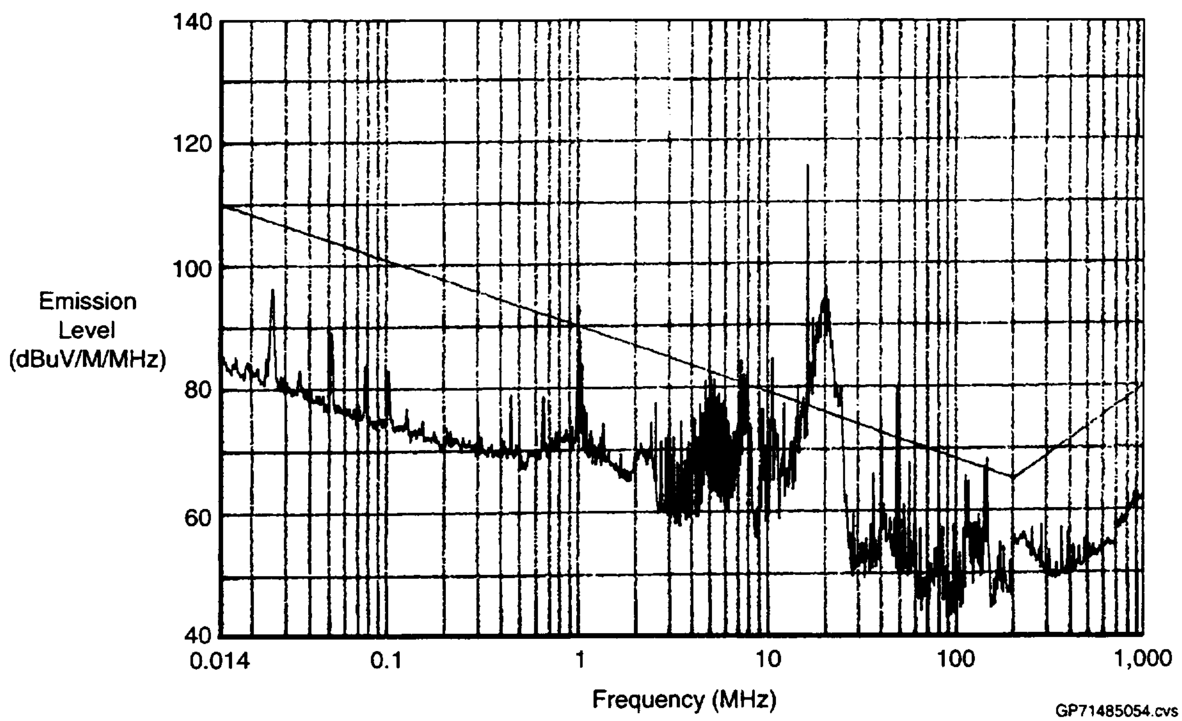


Figure 55. RE02 Horizontally Polarized 0.014 to 1000 MHz Broadband Radiated Emission

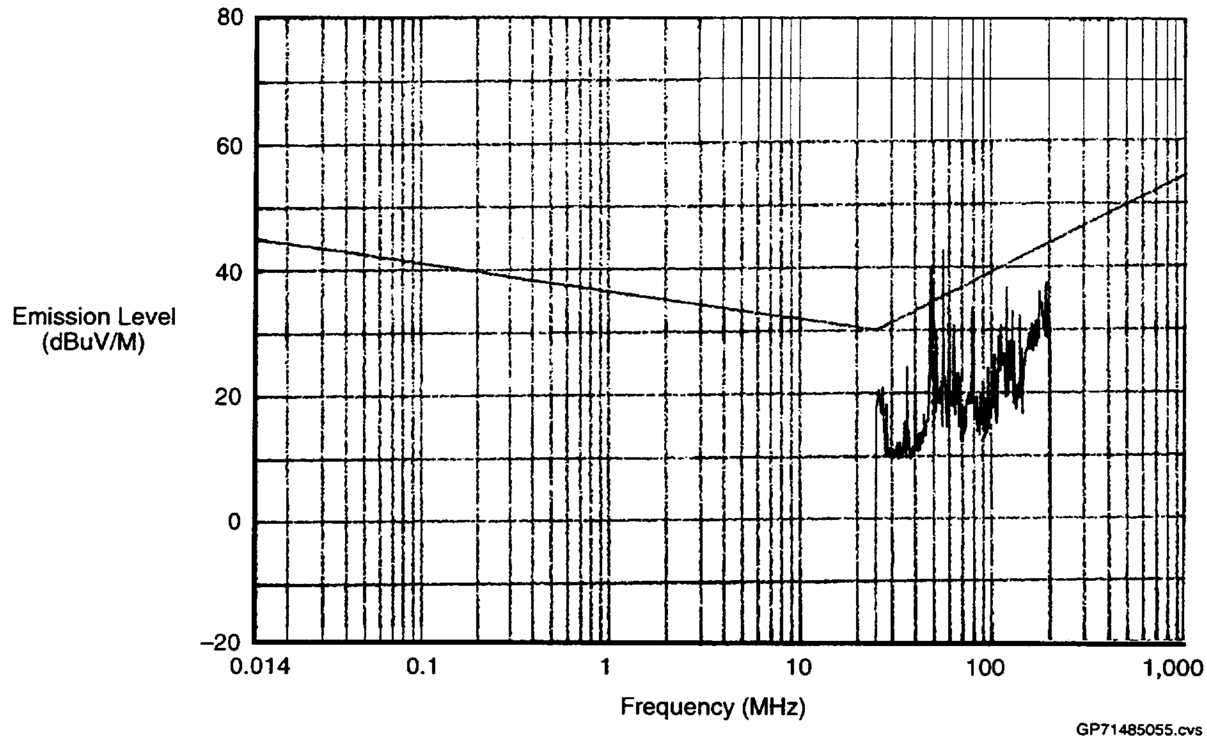


Figure 56. RE02 Vertically Polarized 25 to 200 MHz Narrowband Radiated Emission

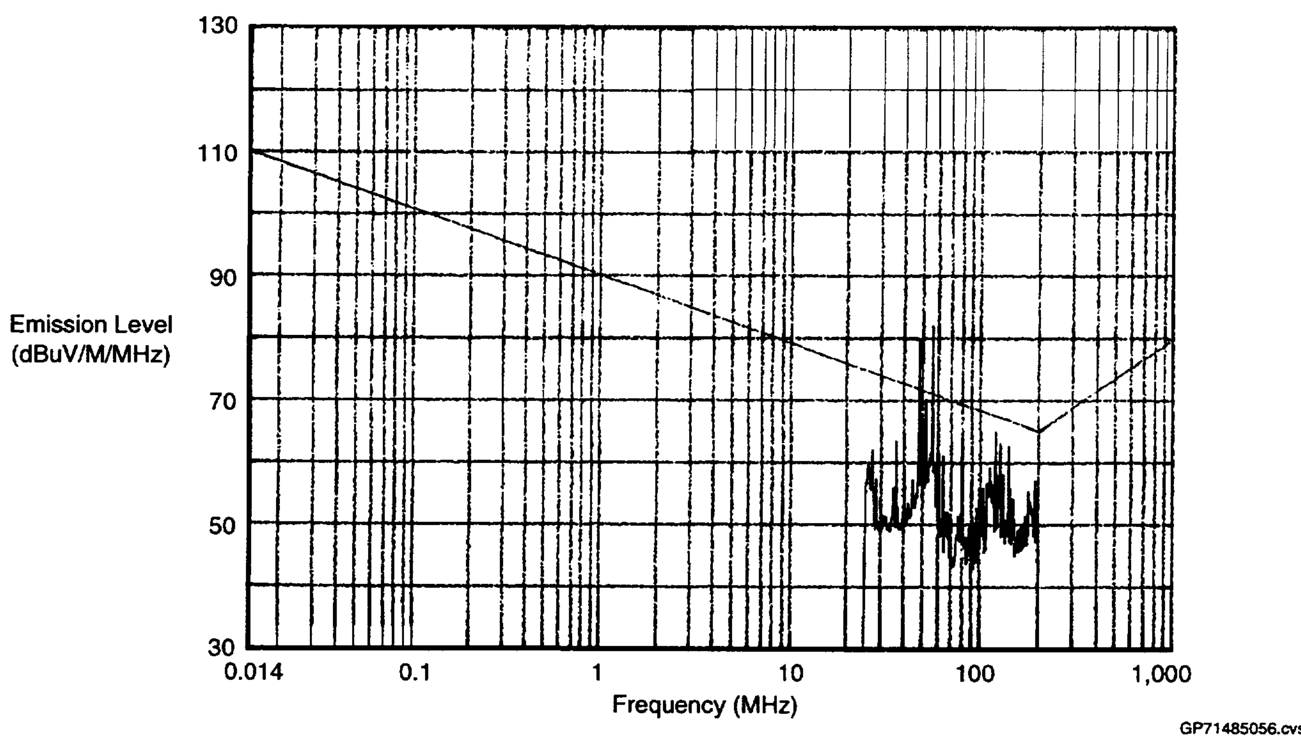


Figure 57. RE02 Vertically Polarized 25 to 200 MHz Broadband Radiated Emission

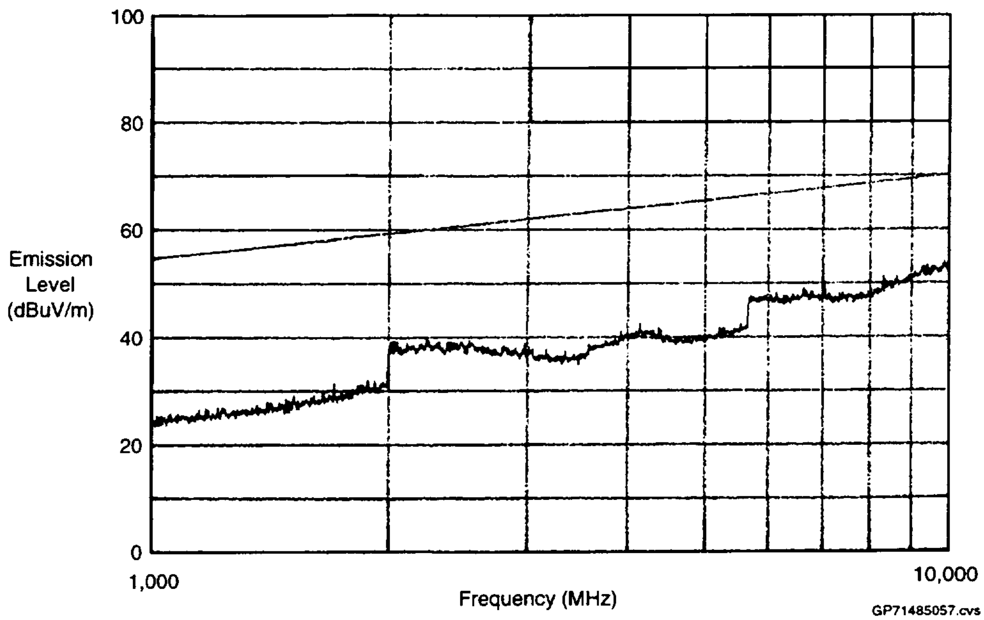


Figure 58. RE02 Horizontally Polarized 1 to 10 GHz Radiated Emission

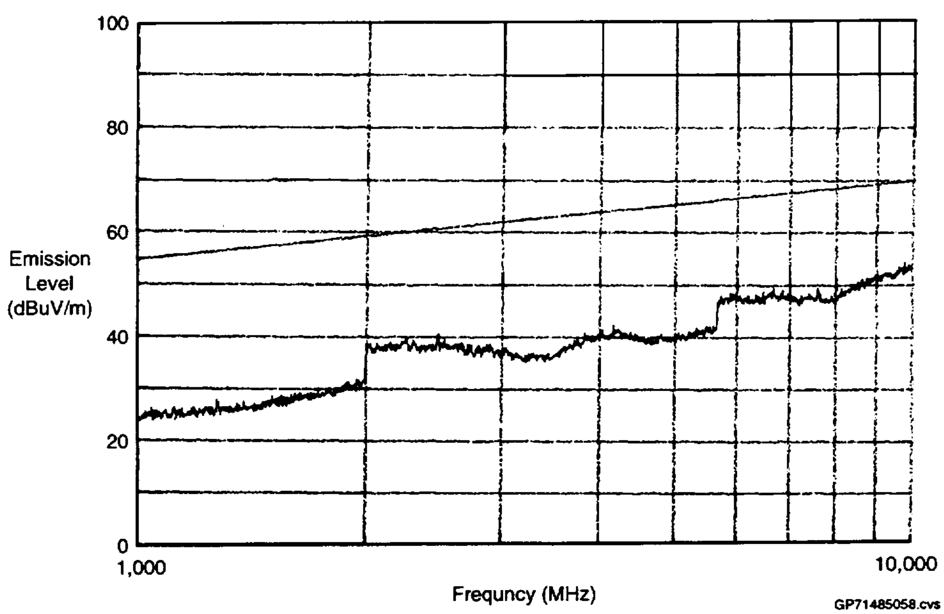


Figure 59. RE02 Vertically Polarized 1 to 10 GHz Radiated Emission

7.8.3.4.7 RS03, Radiated Susceptibility, Electric Fields, 14 kHz – 10 GHz

The FACT system was tested for immunity to radiated electric fields from 14 kHz to 10 GHz at field levels of 200 V/m. The system was tested with both vertically and horizontally polarized fields above 30 MHz. The FACT test setup was very susceptible to these fields in the frequency range of 6 – 500 MHz. The worst case susceptibility was 8 V/m at 178 MHz. It should be noted that the threshold levels stated on the data sheets are the thresholds for the susceptibilities of the FACT system and the support test equipment. Only minimal effort was made to determine whether the observed FACT susceptibilities were actually susceptibilities of the FACT hardware or of the support equipment in the anteroom. This was do to the minimal efforts to RF isolate the unit under test from the support equipment in the anteroom. The effects of the 28 VDC Battery Backup power lines were not tested.

TABLE 18. RS03 RADIATED SUSCEPTIBILITY SUMMARY (PART 1)

Frequency (MHz) [unless noted]	Modulation Frequency/ Percent	Antenna Polarization/ Position	Specification Level (V/m)	Susceptibility Threshold (V/m)	Susceptibility Description
Frequency Range: 14 kHz – 30 MHz					
6.1 – 6.4	80% AM, 400 Hz	Not	200 V/m	180 V/m @ 6.4MHz	FRU Command Current Level Beyond Tolerances
6.4 – 12.4	sine wave	Applicable	200 V/m	10 V/m @ 12.4 MHz	FRU Command Current Level Beyond Tolerances
8.5 – 8.72			200 V/m	180 V/m	Center Tap Monitor Tripped – Valid Data FRU invalid
8.91 – 8.911			200 V/m	180 V/m @ 8.91 MHz	Center Tap Monitor Tripped
9.01 – 9.02			200 V/m	195 V/m @ 8.91 MHz	Center Tap Monitor Tripped
9.11 – 9.31			200 V/m	140 V/m @ 9.18 MHz	Center Tap Monitor Tripped
9.54 – 9.95			200 V/m	150 V/m @ 9.75 MHz	Center Tap Monitor Tripped
10.07 – 12.4			200 V/m	22 V/m @ 12.4 MHz	Center Tap Monitor Tripped
12.4 – 15.4			200 V/m	10 V/m @ 12.4 MHz	FRU Command Current Level Beyond Tolerances
12.4 – 13.65			200 V/m	23 V/m @ 12.4 MHz	Center Tap Monitor Tripped
14.3 – 14.6			200 V/m	185 V/m	Center Tap Monitor Tripped
16.4 – 20.2			200 V/m	15 V/m @ 17.2 MHz	FRU Command Current Level Beyond Tolerances
16.5 – 19.8			200 V/m	28 V/m @ 17.2 MHz	Center Tap Monitor Tripped
20.5 – 30			200 V/m	20 V/m @ 25.8 MHz	FRU Command Current Level Beyond Tolerances
21.5 – 22.3			200 V/m	180 V/m @ 21.8 MHz	Center Tap Monitor Tripped
23.6 – 24.6			200 V/m	130 V/m @ 24.4 MHz	Center Tap Monitor Tripped
25.1 – 25.7			200 V/m	100 V/m @ 25.7 MHz	Center Tap Monitor Tripped
Frequency Range: 30 – 200 MHz with Horizontal Antenna					
34.4 – 44.4	80% AM, 400 Hz sine wave	Horizontal	200 V/m	35 V/m	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances
44.9 – 47.4			200 V/m	105 V/m	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances
47.7 – 48.7			200 V/m	105 V/m	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances
49.8 – 53.5			200 V/m	100 V/m	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances
54.9 – 59.6			200 V/m	50 V/m	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances
64 – 73			200 V/m	120 V/m	Center Tap Monitor Tripped
74.8 – 80			200 V/m	40 V/m	Center Tap Monitor Tripped
			200 V/m	22 V/m	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances
80 – 92.9			200 V/m	Not measured	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances
81 – 92.9			200 V/m	Not measured	Center Tap Monitor Tripped
96 – 105			200 V/m	Not measured	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances
96 – 97			200 V/m	130 V/m	Center Tap Monitor Tripped
103.5 – 105			200 V/m	130 V/m	Center Tap Monitor Tripped
109 – 113			200 V/m	100 V/m	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances Center Tap Monitor Tripped
114.1 – 117.5			200 V/m	Not measured	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances
122 – 130.8			200 V/m	100 V/m	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances
134			200 V/m	140 V/m	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances Center Tap Monitor Tripped
139.5 – 140.6			200 V/m	150 V/m	ICU Command Current Level Beyond Tolerances
141.9 – 143.4			200 V/m	140 V/m	ICU Command Current Level Beyond Tolerances
145.6 – 146.1			200 V/m	Not measured	ICU Command Current Level Beyond Tolerances
149.6 – 179.8			200 V/m	8 V/m @ 178.23 MHz	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances
159 – 160			200 V/m	100 V/m	Center Tap Monitor Tripped
161.5 – 172			200 V/m	100 V/m	Center Tap Monitor Tripped
181 – 190			200 V/m	110 V/m	ICU Command Current Level Beyond Tolerances
190 – 200			200 V/m	110 V/m	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances

TABLE 18. RS03 RADIATED SUSCEPTIBILITY SUMMARY (PART 2)

Frequency (MHz) [unless noted]	Modulation Frequency/ Percent	Antenna Polarization/ Position	Specification Level (V/m)	Susceptibility Threshold (V/m)	Susceptibility Description
Frequency Range: 30 – 200 MHz with Vertical Antenna					
34.5 – 35.2	80% AM, 400 Hz	Vertical	200 V/m	160 V/m	FRU Command Current Level Beyond Tolerances
53.5 – 53.8	sine wave		200 V/m	190 V/m	ICU Command Current Level Beyond Tolerances
63 – 64.4			200 V/m	185 V/m	ICU Command Current Level Beyond Tolerances
67.6 – 68.7			200 V/m	185 V/m	ICU Command Current Level Beyond Tolerances
74.7 – 75.9			200 V/m	160 V/m	FRU Command Current Level Beyond Tolerances
77.6 – 78			200 V/m	200 V/m	ICU Command Current Level Beyond Tolerances
116 – 118.9			200 V/m	170 V/m	ICU Command Current Level Beyond Tolerances
121 – 126.7			200 V/m	100 V/m	FRU Command Current Level Beyond Tolerances
127.8 – 132.5			200 V/m	Not measured	ICU Command Current Level Beyond Tolerances
134.4 – 139			200 V/m	175 V/m	ICU Command Current Level Beyond Tolerances
141.4 – 143.4			200 V/m	160 V/m	ICU Command Current Level Beyond Tolerances
148 – 173.6			200 V/m	60 V/m	FRU Command Current Level Beyond Tolerances
157.9 – 160.2			200 V/m	200 V/m	Center Tap Monitor Tripped
160.9 – 162.9			200 V/m	190 V/m	Center Tap Monitor Tripped
184.9 – 200			200 V/m	100 V/m	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances
Frequency Range: 200 – 500 MHz with Horizontal Antenna					
223.4 – 233.4	80% AM, 400 Hz sine wave	Horizontal Position #1	200 V/m	127 V/m	FRU Command Current Level Beyond Tolerances
260.4 – 280.2			200 V/m	102 V/m	FRU Command Current Level Beyond Tolerances
282 – 287.8			200 V/m	123 V/m	FRU Command Current Level Beyond Tolerances
360 – 382.6			200 V/m	119 V/m	FRU Command Current Level Beyond Tolerances
383.4 – 384.6			200 V/m	136 V/m	FRU Command Current Level Beyond Tolerances
392.2 – 396.4			200 V/m	183 V/m	FRU Command Current Level Beyond Tolerances
426.4 – 433.9	100% AM, 1 kHz square wave	Horizontal Position #1	200 V/m	118 V/m	FRU Command Current Level Beyond Tolerances
435.7 – 439.5			200 V/m	98 V/m	FRU Command Current Level Beyond Tolerances
440.4 – 449.4			200 V/m	94 V/m	FRU Command Current Level Beyond Tolerances
456.3 – 465.6			200 V/m	190 V/m	FRU Command Current Level Beyond Tolerances
225.5 – 233.9	80% AM, 400 Hz sine wave	Horizontal Position #2	200 V/m	146 V/m @ 229 MHz	FRU Command Current Level Beyond Tolerances
260.9 – 290.9			200 V/m	76 V/m @ 272 MHz	FRU Command Current Level Beyond Tolerances
376.1 – 381.2			200 V/m	147 V/m @ 380 MHz	FRU Command Current Level Beyond Tolerances
382.4 – 386.3			200 V/m	165 V/m @ 383 MHz	FRU Command Current Level Beyond Tolerances
451.1 – 460.4	100% AM, 1 kHz square wave	Horizontal Position #2	200 V/m	184 V/m	FRU Command Current Level Beyond Tolerances
469.4 – 496.7			200 V/m	110 V/m @ 475.7 MHz	FRU Command Current Level Beyond Tolerances

TABLE 18. RS03 RADIATED SUSCEPTIBILITY SUMMARY (PART 3)

Frequency (MHz) [unless noted]	Modulation Frequency/ Percent	Antenna Polarization/ Position	Specification Level (V/m)	Susceptibility Threshold (V/m)	Susceptibility Description
			Frequency Range: 200 – 500 MHz with Vertical Antenna		
200 – 210.8	80% AM, 400 Hz sine wave	Vertical Position # 1	200 V/m	95 V/m	FRU Command Current Level Beyond Tolerances
215 – 219.2			200 V/m	145 V/m	FRU Command Current Level Beyond Tolerances
220 – 234			200 V/m	96 V/m	FRU Command Current Level Beyond Tolerances
245 – 278.8			200 V/m	51 V/m	FRU Command Current Level Beyond Tolerances
279.4 – 297			200 V/m	41 V/m	FRU Command Current Level Beyond Tolerances
323.2 – 328.8			200 V/m	121 V/m	FRU Command Current Level Beyond Tolerances
330 – 339			200 V/m	94 V/m	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances Center Tap Monitor Tripped (334 MHz)
358.6 – 382.2			200 V/m	151 V/m	FRU Command Current Level Beyond Tolerances
394 – 397			200 V/m	159 V/m	FRU Command Current Level Beyond Tolerances
450.7 – 458.8	100% AM, 1 kHz square wave	Vertical Position # 1	200 V/m	191 V/m	FRU Command Current Level Beyond Tolerances
469.7 – 496.5			200 V/m	163 V/m	FRU Command Current Level Beyond Tolerances
200 – 211.4	80% AM, 400 Hz sine wave	Vertical Position # 2	200 V/m	112 V/m @ 206.6 MHz	FRU Command Current Level Beyond Tolerances
215 – 219.8			200 V/m	149 V/m @ 218 MHz	FRU Command Current Level Beyond Tolerances
220 – 232.7			200 V/m	130 V/m @ 229.1 MHz	FRU Command Current Level Beyond Tolerances
240 – 275.6			200 V/m	60 V/m @ 255.5 MHz	FRU Command Current Level Beyond Tolerances
278.6 – 296.6			200 V/m	49 V/m @ 286 MHz	FRU Command Current Level Beyond Tolerances
327.2 – 338.3			200 V/m	143 V/m @ 331 MHz	FRU Command Current Level Beyond Tolerances ICU Command Current Level Beyond Tolerances Center Tap Monitor triggered (Rudder FRU Fail)
358.4 – 360			200 V/m	195 V/m @ 360 MHz	FRU Command Current Level Beyond Tolerances
360 – 368.3			200 V/m	150 V/m @ 368 MHz	FRU Command Current Level Beyond Tolerances
370.7 – 381.8			200 V/m	160 V/m @ 377 MHz	FRU Command Current Level Beyond Tolerances
456.5 – 466.1	100% AM, 1 kHz square wave	Vertical Position # 2	200 V/m	135 V/m @ 454 MHz	FRU Command Current Level Beyond Tolerances
			Frequency Range: 500 – 1000 MHz		
500 – 1000		Horizontal	200 V/m		No Susceptibilites Noted
500 – 1000		Vertical	200 V/m		No Susceptibilites Noted
			Frequency Range: 1 – 10 GHz		
1 – 10 GHz		Horizontal	200 V/m		No Susceptibilites Noted
1 – 10 GHz		Vertical	200 V/m		No Susceptibilites Noted

7.8.3.5 EMC Test Conclusion

The FACT system was tested to characterize its EMC performance with regard to the system's airworthiness. The EMC report is intended to only present the test results obtained and is not intended to make or assume any conclusions as to the EMC airworthiness of the FACT system. This data will be presented to NASA for their analysis of the FACT system's airworthiness. The FACT system did not meet all the 461C requirements for CE03, RE02, and RS03 but successfully met the requirements for CE07, CS01, CS02, and CS06.

7.9 Test to be Performed at NASA-Dryden

After the FACT system is delivered to NASA-Dryden, several tests will be performed in the process of preparing the FACT system for flight tests. Acceptance tests will be performed to verify the FACT system works correctly. The FACT system will be integrated with the F/A-18 Iron Bird for validation and verification testing to ensure the system operates as expected and satisfies flight safety requirements. The Iron Bird is integrated with an F/A-18 simulator so the entire FACT fly-by-light and production fly-by-wire flight controls can be exercised and evaluated with a pilot in-the-loop. Aircraft ground testing will be performed to verify the FACT system will operate correctly with the other aircraft systems. These tests are necessary to obtain flight clearance from the NASA Flight Readiness Review Board prior to first flight. During flight tests, the FACT system performance will be monitored during various flight maneuvers and ground maintenance activities. During all tests, the FACT fly-by-light system performance will be compared to the production fly-by-wire system performance.

8. FACT SUMMARY OF RESULTS

8.1 Introduction

The Fly-By-Light Aircraft Closed-Loop Test (FACT) program is a flight test program to demonstrate in-flight optical closed-loop control equivalent to a production Fly-By-Wire system for a rudder control surface. FACT is sponsored by NASA-Lewis Research Center and the Navy's Standard Hardware Acquisition and Reliability Program (SHARP) and flight tested by NASA-Dryden Flight Research Center. Boeing's McDonnell Aircraft and Missile Systems is the system designer, developer, and integrator. This final report describes the FACT system architecture, development, and test up to delivery to NASA-Dryden.

The FACT program was successful at advancing fly-by-light technology for commercial and military aircraft even before the flight test phase. Optic decoding modules, optic sensors, and lessons learned in the FACT program were building blocks used by other programs to advance FBL technology. The Fly-by-Light Advanced Systems Hardware (FLASH) program applied FACT hardware and knowledge to commercial and a variety of military applications to develop near flightworthy fly-by-light components and flight control systems, and the Fly-by-Light Optical Aileron Trim (FLOAT) program applied FACT hardware and knowledge to a commercial transport system. The FACT program intentionally chose a test aircraft that covers the environmental conditions for military and commercial aircraft.

8.2 Development

The FACT program developed avionic interface units, electronic modules, and optic sensors to perform optic closed loop control of a rudder actuator. The Interface Converter Unit (ICU) interfaces the FCC's electrical inputs and outputs to the FACT optical signals. The Feedforward Remote Unit (FRU) converts the feedforward optical command signal into a current command for the actuator's electro-hydraulic valve (EHV). The Electro-Optic Architecture (EOA) module decodes optic sensors. Two Feedforward modules work together to transmit optic command signals from the FCC to the actuator. The Interface module provides most of the ICU interfaces to the aircraft through three independent sections: input power switching, actuator position modulation, and instrumentation interface. Optic position sensors installed in the rudder main ram cylinder use wave division multiplexing and reflective digital code plates to create light patterns that are decoded by the EOA.

Technical challenges were overcome during development of the FACT system components, but problems in building the optical wave division multiplexing components, the optic bricks, for the EOAs and sensors created delays that reduced the program scope. Each optic brick took about one week to make and test, and the process to assemble the optic bricks took about a year to perfect. The delays in producing useable optic bricks delayed the deliveries of EOAs, rudder sensors, and stabilator sensors. The stabilator sensors were delayed so much that there was not enough time left in the FACT program to install the sensors into the actuators, test the actuators, and test the stabilator actuators with the FACT system. The result is the FACT system was designed and developed for a stabilator and rudder flight control surface, but the FACT system was integrated and tested only for the rudder control surface.

The design of the optic brick is an area for improvement. While the optic brick works well and can now be consistently produced, the optic brick contains several parts that are difficult to assemble. Fewer pieces and easier assembly could make optic assemblies with more uniform performance and the possibility of mass production.

8.3 Tests Performed and Results

The FACT equipment integrity and performance was verified for flight test through environmental stress screening to eliminate bad components; component tests to verify ICU and FRU functions; system tests to verify combined FCC, ICU, FRU, and actuator system performance and error handling through the FCC; and environmental airworthiness tests to verify the rudder actuator, ICU, and FRU can withstand the fighter aircraft environment.

8.3.1 Actuator and Optic Sensor Tests

Each rudder actuator passed the slightly modified acceptance test procedure for production actuators. The tests verified the performance of the actuator by testing proof pressure, seal leakage, friction, null position, and sensor output. Before installation into a rudder actuator, each optic sensor was environmentally screened in vibration and temperature tests to provided confidence in the construction of each sensor. One optic sensor passed a lifetime wear test and another optic sensor passed airworthiness tests of vibration, temperature, altitude, and pressure impulse.

8.3.2 Module Environmental Stress Tests

Each module passed vibration and temperature environmental stress screening tests. One capacitor lead on one module broke during these tests resulting in strengthening the capacitor mounting to the module.

8.3.3 Component Tests

The component tests verified the operation of all of the components and functions of the ICUs and FRUs. Each function was tested by varying its inputs and checking its outputs against expected results. After the resolution of the initial problems, the flight control command feedforward and actuator position feedback components performed extremely well in component tests. The components met almost all of the expected results with some minor failures determined to be acceptable. The components operated many hours without failures, showed consistent results when tests were repeated, and showed no unusual or undefined problems.

Throughout the feedforward component tests, the feedforward systems performed solidly. Except for inconsistent but acceptable trip levels for the ICU command versus actuator current, the feedforward system performed as expected and consistently from day to day and from system to system. The feedforward function started operating without fails after power up and reset. The command input versus actual current output was very linear. Failure detection was unfailing; the command versus actual current monitor detected failures, and the feedforward system detected optic signal failures. The system was consistently reset after failures were removed. The optic power margin was very high, about 20 dB. The component test results show the feedforward system is a good system.

Throughout the feedback component tests, the feedback systems performed solidly. Performance was as expected and consistent from day to day and from system to system. The feedback function started operating without fails after power up and reset. The position decoding was accurate. The feedback system detected optic signal failures and was consistently reset after failures were removed. The optic power margin of 5.6 dB to 10.7 dB was adequate. The component test results show the feedback system is a good system.

8.3.4 System Tests

The FACT system performed extremely well when integrated into the flight control system (FCS) with only a few anomalies. The integrated system behaved like the production fly-by-wire system with the FCS on the ground, aircraft weight on wheels, or in the air, aircraft weight off wheels.

Rudder system one failed the maximum deviation from the best fit line in the sensor feedback section of the rudder system performance tests. One data point in the command versus position data deviated from the best fit line by 0.0036 Vrms. The deviation is acceptable since deviation is relatively small at 14%, and the next worse data point deviation is 18 mVrms, 0.007 Vrms under the limit. To make sure no problem exists, NASA-Dryden will retest linearity.

Rudder system two had a couple of acceptable anomalies in the weight on wheels tests. The system needed an extra ICU reset to clear a fault during a power up test and an extra flight control system reset to clear a fault while recovering from a two channel failure.

8.3.5 System Failure Modes and Effects Tests

The integrated FACT and flight control system response to failures was excellent. Failures were always detected and reaction was quick. For the same fail, the FACT feedforward monitor along with the FCC centertap monitor reacted faster than the production system monitor, the FCC monitor with the FACT system reacted the same as the production system monitor, and the FACT feedback EOA software monitor along with the FCC centertap monitor reacted as

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY (*Leave blank*)

2. REPORT DATE
August 1997

3. REPORT TYPE AND DATES COVERED
Final Contractor Report

4. TITLE AND SUBTITLE
Optical Closed-Loop Flight Control Demonstration
Fly-by-Light Aircraft Closed Loop Test (FACT) Program and
Fly-By-Light Installation and Test (FIT) Program

5. FUNDING NUMBERS
WU–538–01–15–00
C–NAS3–25965

6. AUTHOR(S)
Bradley L. Kessler and Michael F. Wanamaker

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
McDonnell Douglas Aerospace
P.O. Box 516
St. Louis, Missouri 63166
and
McDonnell Douglas Aerospace
2401 East Wardlow Road
Long Beach, California 90807

8. PERFORMING ORGANIZATION REPORT NUMBER
E–10860

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135–3191

10. SPONSORING/MONITORING AGENCY REPORT NUMBER
NASA CR–204139
with separate Data Report

11. SUPPLEMENTARY NOTES
This demonstration contains two programs: Fly-By-Light Aircraft Closed Loop Test (FACT) Program by Bradley L. Kessler, McDonnell Douglas Aerospace, P.O. Box 516, St. Louis, Missouri 63166 and Fly-By-Light Installation and Test (FIT) Program by Michael F. Wanamaker, McDonnell Douglas Aerospace, 2401 East Wardlow Road, Long Beach, California 90807. Project Manager, Robert J. Baumbick, Instrumentation and Control Technology Division, NASA Lewis Research Center, organization code 5520, (216) 433–3735.

12a. DISTRIBUTION/AVAILABILITY STATEMENT
Unclassified - Unlimited
Subject Category 06

This publication is available from the NASA Center for AeroSpace Information, (301) 621–0390.

12b. DISTRIBUTION CODE

13. ABSTRACT ***(Maximum 200 words)***

The optical closed-loop flight control demonstration consists of two independent programs. The objective of the Fly-By-Light Aircraft Closed-Loop Test (FACT) program is to demonstrate in-flight optical closed-loop control equivalent to a production Fly-By-Wire system for a rudder control surface. The FACT system has been designed, developed, and tested and is a robust system ready for flight tests on NASA-Dryden's F/A–18 Systems Research Aircraft. The FACT program was sponsored by NASA-Lewis Research Center and developed by McDonnell Douglas. The FACT architecture inserts electro-optic avionics between the flight control computer and a rudder actuator modified with an optic sensor. The avionics transmits optic command signals for the actuator, determines actuator position by decoding the optic sensor, and provides error monitoring. Several tests verified the system's readiness for flight test. Component tests verified proper operation of each low-level function. Vibration, temperature and altitude, and electromagnetic compatibility tests verified the system can survive the military aircraft environment. System tests verified FACT system performance was equivalent to production system performance. This final report describes the FACT system architecture, development, and test up to delivery to NASA-Dryden. Also described is the Fly-By-Light Installation and Test (FIT) program that investigated optic fiber installation, repair, and signal quality.

14. SUBJECT TERMS
Fly-by-Light; Control with photonics; Fiber optics; Optical signalling; Optical sensors

15. NUMBER OF PAGES
88

16. PRICE CODE
A05

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	